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PROPULSION REQUIREMENTS

FOR SPACE MISSIONS

VOLUME I

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ROCKETDYNE

A DIVISION OF NORTH AMERICAN AVIATION, INC.

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The many studies of space propulsion systems conducted previous to this effort both at Rocketdyne and in the open literature have provided an excellent background. These contributions have facilitated these studies and are greatly appreciated.

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FOREWORD

This report was prepared in compliance with the provisions of National Aeronautics and Space Administration contract NAS 5-916, "Research Study to Determine Propulsion Requirements and Systems for Space Missions."

ABSTRACT

Volume I is a summary of the results and recommendations of a "Research Study to Determine Propulsion Requirements and Systems for Space Missions." The investigation was directed toward characterizing propulsion systems for future space missions, providing preliminary data and optimization criteria, and determining specific propulsion areas which require future investigation.

(Unclassified Abstract)

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INTRODUCTION

The "Research Study to Determine Propulsion Requirement and Systems for Space Missions" (NASA Contract NAS 5-916) was initiated to provide a broad evaluation of spacecraft propulsion requirements. This eight month investigation was directed toward characterizing propulsion systems for future space missions, providing preliminary propulsion data and optimization criteria, and determining specific propulsion areas which require future investigation.

The determination and the evaluation of design parameters for space propulsion demands a knowledge of the over-all space mission. The space mission and space environment are inherent factors in the design of a space propulsion system. The engine's function in the proposed space missions, and alternate applications, must be analyzed to determine optimum designs. Thus mission/trajectory analysis is an area of major analytical effort in the contracted investigation.

Alternate, and widely separated approaches for a mission analysis investigation can be used: (1) All possible factors affecting propulsion can be analyzed for each mission; an extensive, time consuming effort that may never lead to engine design or

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development; (2) or a minimum effort approach can be undertaken simply based on personal (and possibly prejudiced) concepts in propulsion requirements. Design and development could be initiated immediately, but later modifications to make the system usable in the various space missions would probably be required.

In this space propulsion investigation a realistic approach which is between the two extremes has been followed to provide the best possible information within the contract period. Previous experience in engine and system optimizations provided a knowledgable selection of what criteria affect engine design and performance, and the general methods of propulsion analysis.

Where past studies have indicated trends these results are employed and supplemented with additional analyses.

Since the investigation has been directed to be a broad establishment of propulsion requirements, the intent was not to establish detailed propulsion design.

Technology advances may change some of the conclusions; however, an attempt has been made to anticipate advancements in design technology (lower weights, improved insulation, etc.).

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In evaluating propulsion systems, the requirements specified have not been subjected to extensive influence by current trends or capabilities; that is, propulsion systems needed immediately for an initial space mission project and governed by current structural, guidance, and subsystem designs have been somewhat divorced from this study, since the purpose was to establish propulsion requirements for future optimum system designs.

The study has been divided by effort, and reporting, into two phases. The first phase consisted of generalized mission and trajectory maneuver analysis, together with a preliminary propulsion analysis. The second phase effort was directed to analyze certain recommended space systems in further detail. The detailed descriptions of the study effort and results are presented in Volume 2 (Phase 1) and Volume 3 (Phase 2) of this report. The results and conclusions from these studies are summarized and presented in this volume (Volume 1).

INVESTIGATION SCOPE AND PHILOSOPHY

A brief outline of the manner in which the two phases of the study were conducted is presented in the flow charts, Table 1-1 and 1-2. As indicated, Phase 1 was begun with a comprehensive review of the literature concerned with space missions and propulsion. From this review, realistic space missions were selected for consideration in the study. These selected missions are composed of a variety of propulsive maneuvers, many of which are common to several space missions. To facilitate the analysis, these maneuvers were outlined, and the propulsive analyses conducted on a maneuver basis. Throughout this maneuver analysis, the literature was scrutinized for useful information.

Following the extensive analysis of space propulsive maneuvers, these maneuvers were recombined into the selected space missions. A review of current and proposed booster systems was initiated and basic vehicles were selected as applicable for future space missions. A preliminary investigation of space vehicle stages for these boosters was conducted to indicate performance in the selected space missions. Various propulsion systems (liquid propellant, solid propellant, thermo-nuclear, ion-electrical, etc.) were then considered for the

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F... STUDY OUTLINE

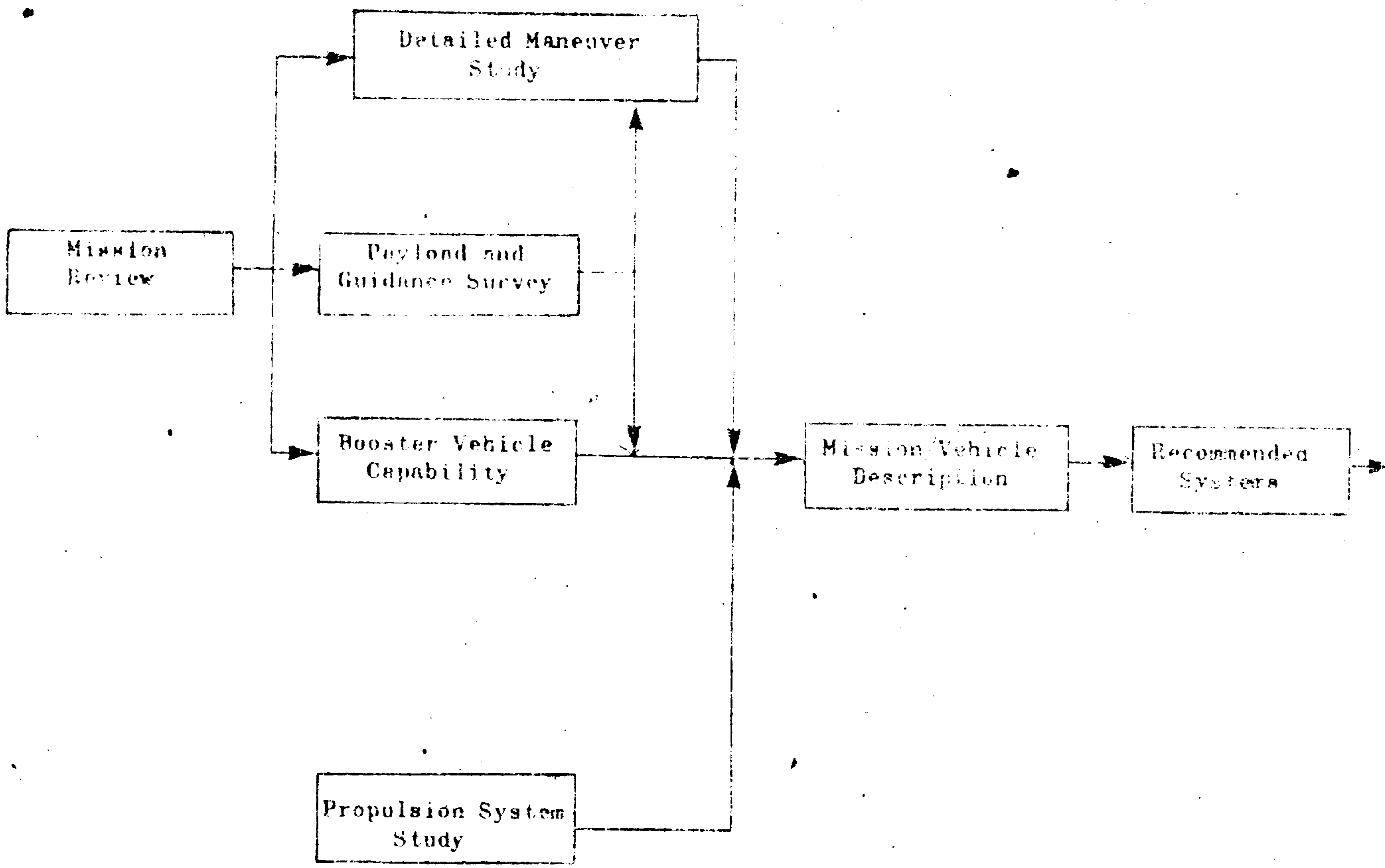

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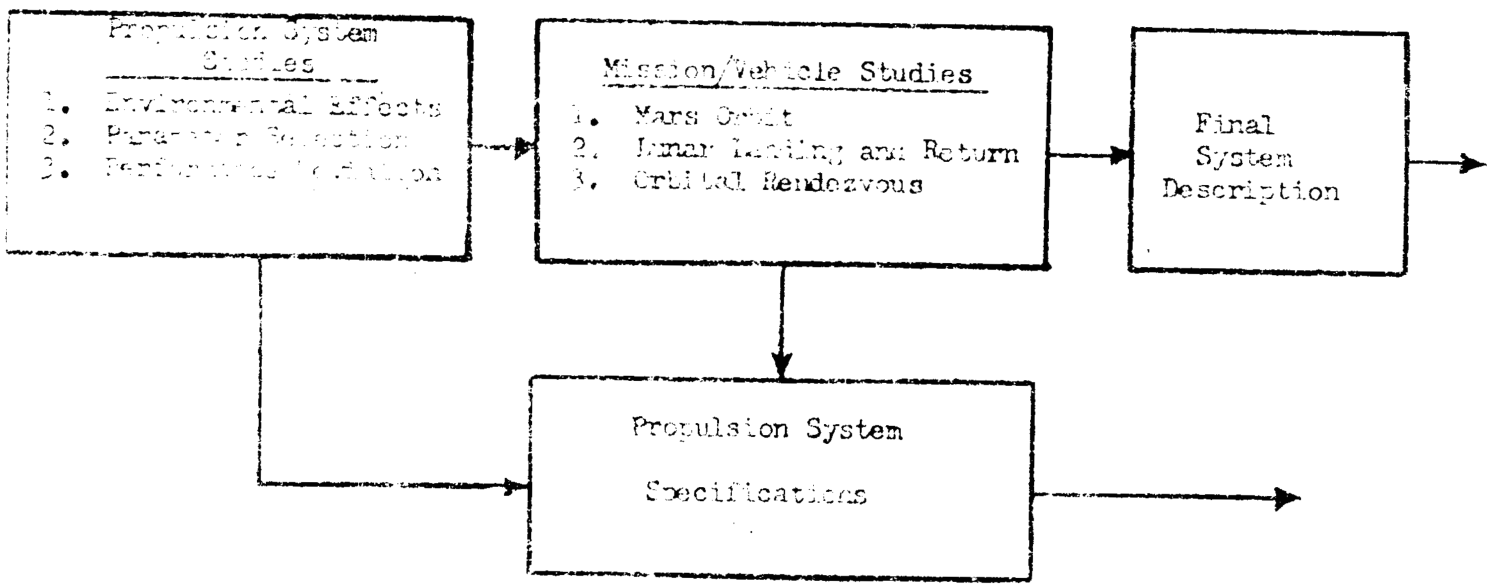
TABLE 1-2

PHASE II STUDY EFFORT



space stages and evaluated regarding their utility in the mission. Based on the performance, and review of space payload requirements, promising liquid-propellant propulsion systems that would provide a useful payload with a basic booster vehicle were recommended for further study.

The Phase 2 effort was directed to consider three space missions: soft lunar landing and return, Mars orbit establishment, and earth orbital rendezvous. The latter mission was to be considered secondary in importance.

A variety of methods of accomplishing each space mission trajectory was considered, and the most desirable method selected in a continuation of the Phase 1 trajectory/maneuver studies. Space vehicle systems to achieve the mission were investigated and described in terms of staging and propulsion requirements. The propulsion systems to be used in the space missions were analyzed in considerable detail regarding such parameters as propellants, feed systems type, etc. Effects of the space environment, operating parameter selection, and transient operation effects were considered.

Based on the investigation results a list of specifications describing the propulsion system requirement that should be considered in selecting and evaluating a propulsion system for a space mission was developed. For the three space missions, the specification listing was also compiled for the selected propulsion systems. From the effort conducted, recommended future investigations necessary to completely characterize the space mission and propulsion system have been formulated and listed.

SUMMARY

The following items briefly summarize the results of the study.

I. Propulsion System - A number of propulsion system studies were conducted. The conclusions of these investigations are listed below.

A. High-energy, liquid propellant systems offer significant performance advantages over other combinations. Upon their further development, more advanced systems such as nuclear and ion will offer tremendous performance advantages over the more conventional liquid propellant systems.

B. The environmental factor exerting the most significant effect on the space propulsion system is heat transfer leading to propellant heating.

C. For long periods of storage in space, the heat transfer internal to the propulsion system (conduction through structure, etc.) may lead to excessive propellant heating and in conjunction with external heating limit the use of high-energy, cryogenic propellants.

D. Use of a fuel loading bias is advantageous in reducing trapped propellants particularly in the cryogenic (LO_2/LH_2) systems.

E. Space propulsion systems supplying large amounts of energy may use propellant utilization systems to advantage in reducing trapped propellants.

F. Thrust vector control requirements are small but necessary.

G. Thrust chamber performance losses due to throttling are slight.

H. Variation in space vehicle velocity due to variations in main engine cutoff impulse are on the order of 1 fps or less.

II. Lunar Landing and Return Mission - This mission was investigated based on a space vehicle placed in an Earth orbit by a NOV[illegible] booster vehicle. The conclusions below are generally based on this space vehicle whose gross weight is 354,000 lb.

A. Two methods of accomplishing the lunar mission are generally available:

1. Direct lunar landing

2. Lunar landimg via an intermediate orbit. The latter method is generally recommended for manned missions.

B. Two-stage vehicles are recommended for both methods. For the direct landing, staging occurs at lunar surface. For the intermediate orbit method, staging occurs in the lunar orbit.

C. Oxygen/hydrogen, pump-fed propulsion systems are recommended for all space stages for both methods.

D. The first-stage engine for the vehicle using an intermediate orbit should have a thrust in the 100,000 to 200,000-lb thrust range. The optimum thrust is 125,000 lb thrust or a thrust-to-Earth weight of about 0.35.

E. The second-stage engine for the vehicle using an intermediate orbit should have a thrust of about 77,000 lb thrust. (Earth-thrust-to-weight ratio of 0.68.) This thrust level depends on the height of the second intermediate orbit. The propulsion system should have the capability of 6:1 step throttling and 6 percent continuous throttling.

F. For the direct lunar landing the first stage should have a thrust of 200,000 to 300,000 lb thrust (Earth thrust-to-weight ratio of 0.

[redacted]

G. The second stage of the direct landing vehicle should have a thrust of 40,000 to 80,000 lb thrust (Earth thrust-to-weight ratio of 1.5).

H. In most of the propulsive maneuvers for the lunar mission the thrust level has only slight effect on the payload capability. Operation at off-optimum conditions does not unduly affect the payload capability.

I. Space vehicle size has little effect on the propulsive maneuvers used in accomplishing the lunar mission. For small vehicles, however, a manned mission would be unfeasible, and the direct landing approach might be selected over the intermediate orbit method.

J. Space vehicle size should not influence the use of LO_2/LH_2 propellants to any great extent. For the relatively short times involved in the lunar mission (1 to 2 weeks) cryogenic propellants could be maintained even in fairly small vehicles.

K. Thrust level can be scaled according to the vehicle gross weight.

L. All vehicles are assumed to use aerodynamic re-entry for returning to Earth.

III. Mars Orbit Establishment - This mission was investigated based on a space vehicle placed in an Earth orbit by a NOVA H-6* booster vehicle. The conclusions below are generally based on this space vehicle whose gross weight is 354,000 lb.

A. The Mars mission consisted of departure from an Earth orbit and simultaneous plane change. A preliminary orbit is

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established around Mars. This orbit is corrected to the desired 300 n mi orbit.

B. Transfer times on the order of 200 days are considered.

C. A two-stage vehicle is recommended. The first stage accomplishes the Earth departure maneuver. The second stage establishes an orbit about Mars and provides corrections to establish the 300 n mi orbit.

D. For this large vehicle the oxygen/hydrogen propellant combination is recommended for both stages. Some propellant storage problems may occur in the second stage. Further and more detailed investigations of this problem may indicate that a storable propellant system is warranted.

E. Since the energy requirements vary with the launch date, it is recommended that the Mars vehicle be designed with the propellant capacity for lahuching over an interval of dates. Although decreasing the payload capability, this provides a realistic, flexible design approach.

F. The first-stage engine has a thrust of 150,000 lb (Earth thrust-to-weight ratio of 0.50).

G. Second-stage thrust was 30,000 lb.

H. Thrust level has but slight effect on payload capability. Operation of off-optimum conditions does not unduly affect the payload capability.

I. Space vehicle size has little effect on the propulsive maneuvers used in accomplishing the mission.

J. Thrust level can be generally scaled according to vehicle size.

K. Vehicle size should not influence the use of high-energy propellants in the first stage.

L. The use of cryogenics in the second stage is definitely affected by vehicle size. Use of LO_2/LH_2 in vehicles smaller than that considered might be unattractive due to the extensive insulation requirements. More detailed analyses of the heating problem should be considered.

IV. Earth Orbital Rendezvous

A. In the rendezvous mission, the propulsion system parameters (thrust specific impulse, etc.) do not significantly affect performance.

B. Mission operational aspects are important and significantly affect the mission maneuver description.

C. For the selected rendezvous mission involving a 5 degree plane change requirement, the rendezvoused payload is considerably reduced from that payload ordinarily placed in a 300 n mi.

DESCRIPTION OF RESULTS

VEHICLE REQUIREMENTS

A review of the requirements for manned space missions has indicated the necessity for large booster systems; or of a successful system for orbital rendezvous and space stage buildup for manned space missions.

An investigation of manned space capsule requirements has been conducted and the resultant space capsule weights have been summarized in Fig. 1-1 (less propulsion system weight). For a manned lunar mission, (3 men and approximately 10 days duration) an earth return capsule weight of approximately 30,000 lb is required based on current assumptions for adequate Van Allen and solar radiation shielding. Of the 30,000 lb, 17,000 lb is radiation shielding, and 6,000 lb is for short-transportation, taxi capsules. If these items were omitted, the space capsule weight would be only 7000 lb. If a system for forecasting solar flares can be devised, a major portion of the 17,000 lb for radiation shielding could be eliminated.

In Table 1-3 are presented the preliminary estimates of space mission payload capabilities. Analysis of these payloads (Table 1-4) indicate that booster vehicles larger than the Atlas and Saturn C-1 systems should be considered in order to present a comprehensive investigation of space missions. Thus, the Nova systems have been included in the analysis so that the effect of

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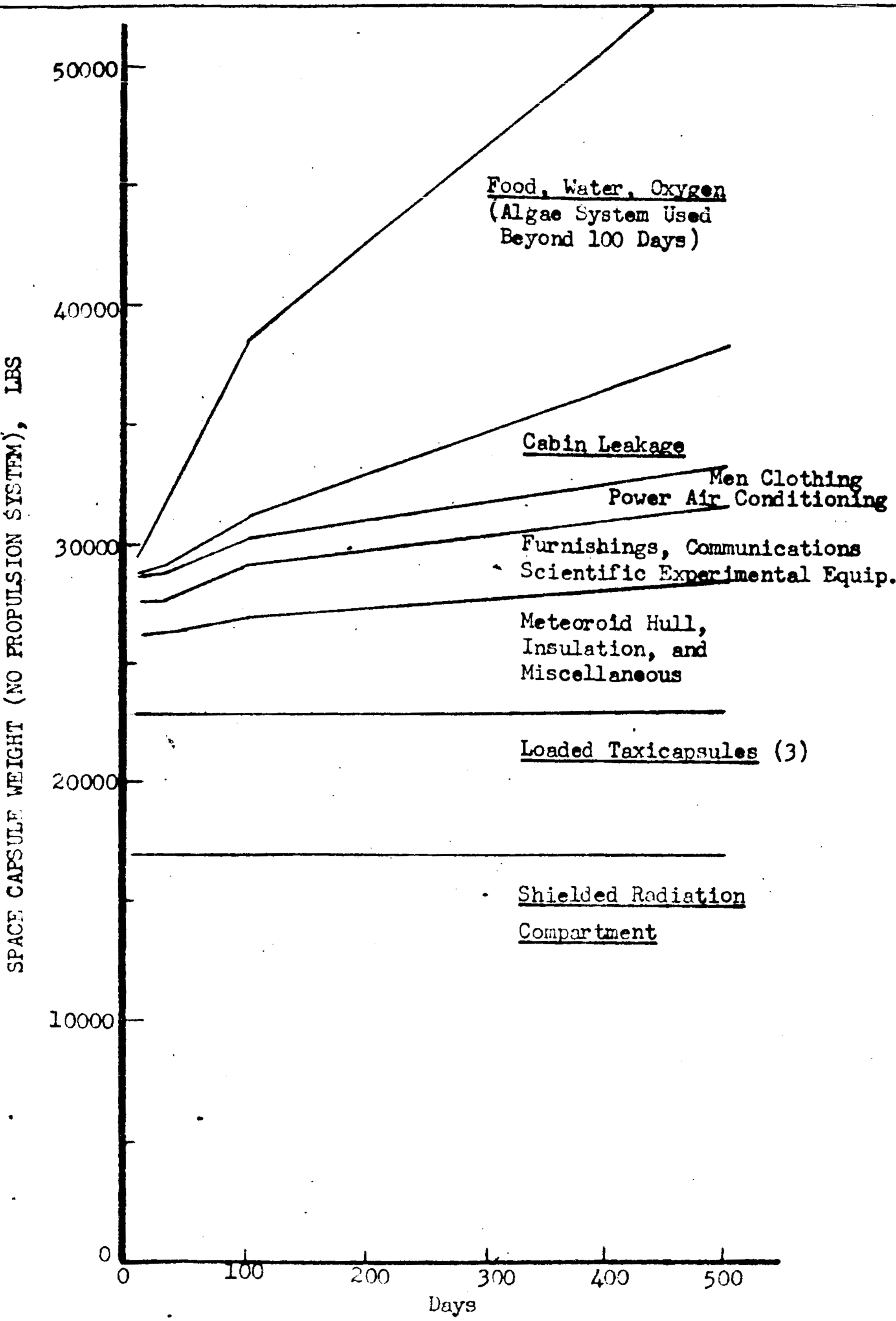

Figure 1-1 . Three-Man Space Capsule, No Propulsion System

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TABLE I

PAYLOAD CAPABILITY FOR SPACE MISSIONS

VEHICLE	300 N. Mi. Orbit	24 Hour Orbit	Soft Lunar Landing	Lunar Satel. & Return to Earth's Surface (Aerodynamic)	Lun & R Ear (Ae
Atlas Centaur	8,500	2,600	1,750	1,370	
Saturn C-1	24,300	7,500	5,000	3,900	1,
Saturn C-2	50,630	15,600	10,300	8,100	2,
Saturn Nuclear	68,200	24,300	15,200	12,400	4,
Nova R-2	73,000	22,500	15,000	11,700	4,
Nova H-2	118,000	33,300	24,000	18,800	6
Nova R-4	146,000	45,000	30,000	23,400	8
Nova H-4	236,000	72,700	48,000	38,100	13
Nova H-6	354,000	109,000	72,000	56,500	19
Nova N-4	505,000	198,800	126,000	101,000	35

Pc
N
Pa

* A LO_2/LH_2 Pump-Fed Propulsion System is used for Space Stages.

B For the Nova class vehicles, the letter represents the second stage propellant: R- RP-1, H-hydrogen, N - nuclear. The number designates the number of F-1's clustered in the first stage of the Nova vehicle.

- 3

BASED ON CURRENT BOOSTER VEHICLES

ar Landing ,eturn to th's Surface rodynamic)	Lunar Landing & Return to Earth Orbit	Mars Satel. & Return to Earth Orbit	Soft Mars Landing	Venus Satel. & Return to Earth Orbit	Soft Venus Landing (Non-Aerodynamic)
450	250	190	469	175	165
350	800	540	1,330	500	480
800	1,650	1,120	3,590	1,050	,995
160	2,450	1,580	3,940	1,870	1,775
100	2,350	1,610	4,000	1,510	1,435
,700	3,850	2,600	6,490	2,440	2,320
,300	4,750	3,220	8,000	3,020	2,870
,300	7,650	5,200	13,000	4,880	4,640
,900	11,500	7,800	19,400	7,340	6,950
,000	20,000	13,400	32,600	14,650	13,900

tential
anned
yloads

Unmanned
Payloads

[redacted]

TABLE 1-4

SPACE PROPULSION SYSTEM DESCRIPTION

Propulsion System: LO_2/LH_2 Pump-Fed

Manned Mission	Payload Returned, lb 3 Man	Payload Returned, lb 6 Man	Gross Wt. in 300-n mi Orbit, lb 3 Man	Gross Wt. in 300-n mi Orbit, lb 6 Man	Typical Boost Vehicles Required 3 Man	Typical Boost Vehicles Required 6 Man
Lunar landing and return to Earth [illegible] [illegible] [illegible]	[illegible]	......	[illegible]	......	11 Saturn C-2, [illegible] Nova H-2, [illegible] Nova H-4	
[illegible]	[illegible]	......	224,[illegible]	......	9 Saturn C-1, 5 Saturn C-2, 1 Nova H-4	
Mars orbit and return to Earth [illegible]	[illegible]	[illegible]	[illegible],000	5,090,000	[illegible] Nova H-6, 13 Nova H-4, 25 Nova H-2	14 Nova H-6, 22 Nova H-4, 43 Nova H-2
Venus [illegible] return to Earth [illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible] Nova H-6, 11 Nova H-4, [illegible] Nova H-4, [illegible] Saturn C-2	11 Nova H-6, 21 Nova H-4, 42 Nova H-2

large payloads on the design of space propulsion systems can be studied. The payload capabilities presented are preliminary, since many of the vehicles are not designed or built, but the data do accurately indicate the trend in space mission capabilities.

For a manned Mars orbital and return space capsule, an approximate 70,000 lb system was found necessary based on the shielding assumptions. A four stage liquid-chemical propulsion system (with 70,000 lb space capsule) would require approximately a three million lb space vehicle initially in an Earth-orbit; a two stage nuclear propulsion vehicle could reduce the gross weight in orbit to approximately 850,000 lb; and an ion propulsion space vehicle, to approximately 175,000 lb. Thus, if the liquid-chemical system were employed, orbital buildup would be required with even the largest NOVA systems considered. With the ion system, a single H-6 could place the space vehicle in the starting orbit.

PROPULSION SYSTEMS FOR SPACE

A variety of propulsion system types were considered for application in space missions. These systems are compared on the basis of available specific impulse and characteristic thrust-to-weight ratios in Fig. 1-2.

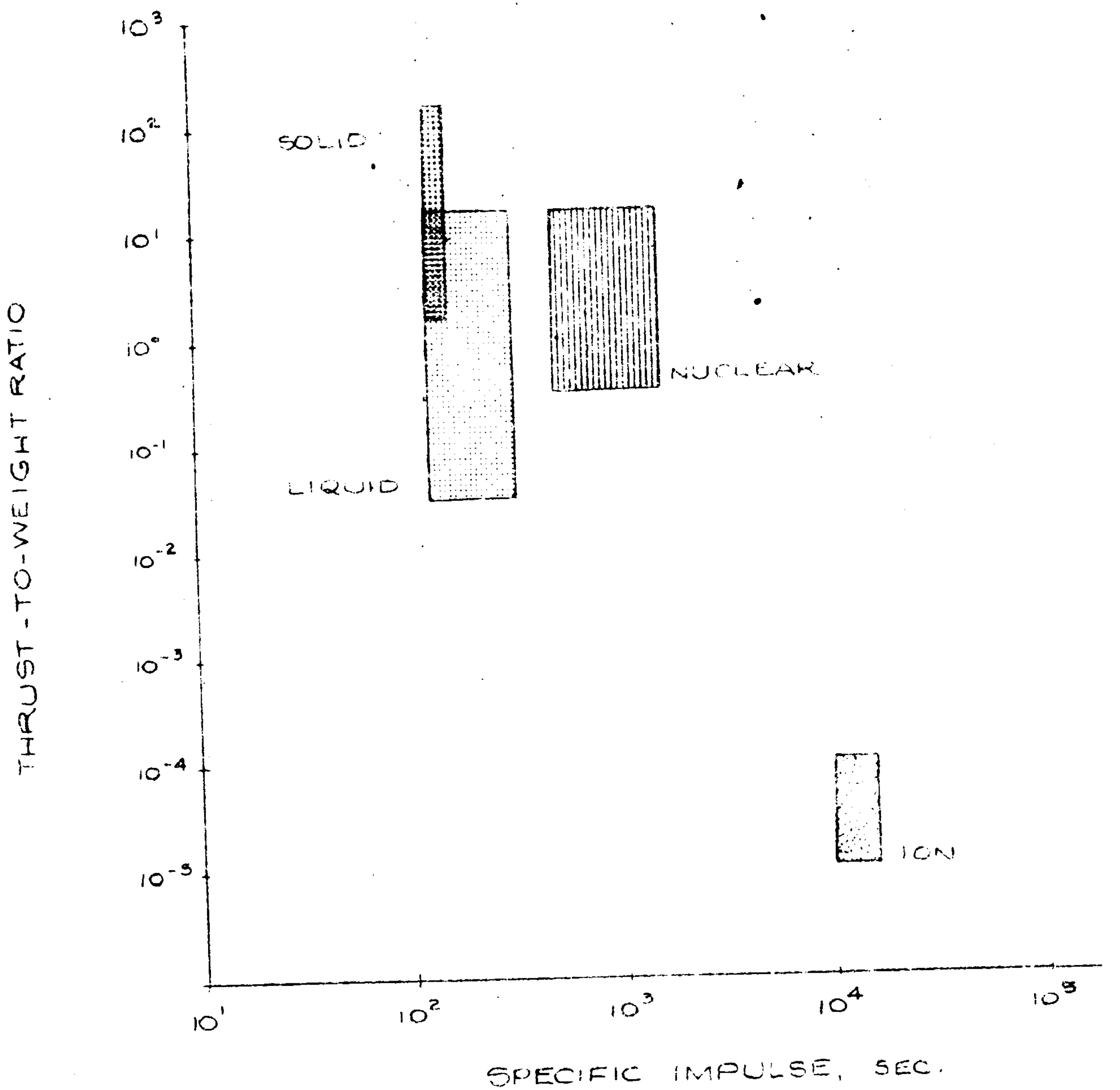

Figure 1-2. Comparison of Space Propulsion Systems

Liquid Propellant Propulsion Systems

Review of the space propulsion requirements indicated that the high-energy, cryogenic, liquid-propellant systems have a performance advantage over "Earth-storable" liquid systems. Since the velocity requirements for most space maneuvers are large, the high-energy cryogenic systems would be recommended for almost all applications. Only at the low ΔV applications, or where some consideration other than performance was influential, would an "earth-storable" system compete.

Consideration of the storage problems associated with cryogenic propellants indicates that the severe propellant heating occurring in some missions would necessitate such a large amount of insulation for the cryogenics that the storable propellant systems would provide a larger payload.

In missions requiring extremely long storage times (return from Mars missions) the earth storable propellants may be used to advantage. This is discussed further in a later section.

Solid Propellant Propulsion Systems

The relatively low specific impulse associated with most solid propellant systems limits their applicability for space propulsion maneuvers. The higher performance liquid systems provide a significant payload advantage.

The high thrust-to-weight ratios associated with solid propellant motors are not required for most space maneuvers. The liquid system, designed for a low thrust-to-weight provides a reduction in inert weight that further enhances their comparative performance. This payload advantage increases as mission velocity requirements increase. Only for space propulsion maneuvers requiring less than 1000 to 2000 ft/sec do solid propellant systems appear to be comparative unless a high thrust-to-weight ratio (inherent in the solid propellant designs) is required. An abort escape rocket system may be such an example.

Nuclear Propulsion

Performance advantages of a nuclear propulsion system for orbital escape maneuvers indicate these systems will find definite applications in this area in future space missions. Suborbital flight with nuclear propulsion requires a close review of possible hazards associated with nuclear engine operation. The development schedule for nuclear engines limits their immediate applicability for space propulsion.

Ion-Electrical Propulsion Systems

Review of performance requirements for manned interplanetary flight tends to indicate that ion propulsion systems will be required for such missions.

These systems, with their inherent high specific impulse (8000 to 12,000 sec) will provide the high-weight manned payload return capsules with a feasible initial space vehicle weight. The ion system has two basic limitations: first, the Earth booster and the planetary landing missions require an additional high thrust-to-weight propulsion system for these propulsion phases: second, ion thrust levels required for manned space missions are approximately 10 to 20 lb; this far exceeds the current development status for these engine systems.

The ion-electrical propulsion system also finds application for attitude control of earth orbital or interplanetary space vehicles. Low thrust-to-weights are required in this application, making the ion system a prime contender because of its high specific impulse, This factor is further enhanced if an electrical power source, such as a nuclear reactor system, is required for communication and other system power requirements.

PROPULSIVE MANEUVER DESCRIPTIONS

The propulsive maneuvers used in accomplishing space missions can be performed in a number of different ways. In the study of many of these maneuvers, certain methods of performing them appeared more advantageous than others. Other maneuvers, such as the landing maneuver, possess

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myrid possibilities and it is difficult to select a single method as the most advantageous at the present time. To provide a concise description of the various maneuvers necessary for space flights, a brief discussion and summary is provided. The general philosophy of accomplishing each maneuver is explained and methods holding particular advantage (if any exist) denoted.

Earth Vicinity Maneuvers

The following maneuvers are those that generally occur in the vicinity of the earth, being initiated at the earth's surface or in a low altitude earth orbit.

Orbital Rendezvous. This maneuver was studied considering two vehicles initially separated due to launch inaccuracies, to rendezvous in a 300 n. mile parking orbit. On the basis of minimum fuel requirements the method of accomplishing the gross changes of this maneuver consisted of a plane change at node with an eccentricity change for circularization of the orbit at apogee or perigee (depending upon the change in the major axis) to establish the 300 n. mile circular orbit. Terminal phase of the maneuver considered a pulsed thrust trajectory resulting in contact of the two vehicles. Thrust-to-weight ratios in the range 0.05 to 0.1 appear

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promising for this maneuver. Velocity requirements for rendezvous are on the order of 2800 fps. Some 2200 of this is required by the plane change. Ion systems do not appear too feasible for short time rendezvous applications. The solid systems have a relatively low I_s and the multiple start capability required for most rendezvous missions appear to limit the application of solid propellant systems.

The analysis conducted indicates that in most propulsion phases the small velocity requirements necessary would not dictate a nuclear or high-thrust liquid chemical system. A small pressure-fed propulsion system would comply with the propulsion requirements. Reignition of the booster phase, upper-stage engines appears to be unwarranted for the rendezvous application.

Planetary Transfer. The transfer trajectory maneuvers for relatively high thrust vehicles were studied on the basis of patched conic sections. These conic sections representing the geocentric, heliocentric, and planetocentric phases of operation are patched together to provide a simple, accurate simulation of the actual trajectory.

The total velocity requirement for a planetary transfer is a combination of several velocity increments: launch, plane change, mid-course correction, capture, etc. The capture phase may be either direct landing or orbit

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establishment. If plane change occurs at launch only two major propulsion phases will be required, an earth escape phase, and a capture or planetary phase.

The velocity requirements for these phases vary greatly with mission and, for each mission, with launch date and transfer time. Practically, it is advisable to design systems applicable to a range of launch dates. Two stages are designed to have maximum propellant capacity sufficient to meet the velocity requirements for any date of the given range of launch dates. For any date in this period, the tanks of each stage are loaded to the capacity required for the given launch date.

Planetary transfers to Mars and Venus were considered. From the standpoint of energy requirements minimum transfer times (high thrust-to-weight ratios) appear to be in the vicinity of 180-200 days and 100-110 days for Mars and Venus respectively. Analysis of Earth phase propulsion for a lunar or interplanetary mission indicates that the optimum liquid propellant propulsion thrust-to-weight ratios are in the 0.3 to 0.8 range. The stage velocity increment required for this phase range from approximately 10,000 to 20,000 ft/sec based on initiation from a low altitude Earth orbit. Review of performance requirements for the earth escape propulsion indicate that a high-energy cryogenic liquid propellant system would be recommended. To facilitate further analysis of orbit initiated earth

[redacted]

escape phase, a nomograph indicating the effect of mission on the propulsion system has been formulated. Low thrust-to-weight (electrical) systems appear to have some advantages in the performance of interplanetary missions. Thrust-to-weight ratios on the order of 10^{-4} appear to be a minimum value. A considerable reduction in the initial system weight is possible using advanced electrical propulsion systems.

Lunar Transfer. The earth phase maneuver of a lunar transfer has the requirement of raising the vehicle energy to the level necessary to enter a selected earth-moon coast trajectory. This maneuver was considered to initiate in a low altitude earth satellite orbit which allows a greater latitude in launch times, although slightly less efficient than a direct earth launch. The trajectory for this maneuver is based on a tangential thrust (thrust aligned with velocity) flight path until the desired energy level for the geocentric earth-to-moon coast trajectory is attained. Selection of this thrust attitude method is based on minimum velocity requirement consideration.

The earth-moon transfer trajectory is in the form of geocentric conic sections, the type of which depends on trip time. The various trip times require different energy levels. The energy level of the transfer

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trajectory varies inversely with trip time. From energy level considerations and Lunar retro-thrust requirements, trip times between 2 to 3.5 days appear attractive. The deviations in Earth escape phase velocity increments resulting from wide variations in launch time and Earth/moon coast trip time are extremely small. Thus, a single system design for this Earth escape phase would have wide applicability.

Desirable thrust-to-weight ratios for this maneuver are in the 0.1 to 0.8 range. Velocity requirements for the transfer are on the order of 10 - 11,000 fps.

Twenty-Four Hour Orbit Establishment. Study of the establishment of a 24 hour stationary orbit was conducted considering the system initially in a low altitude orbit whose plane is inclined 28.8 degrees to the equatorial plane (AMR launch). The vehicle coasts in this low altitude orbit until the desired spatial positions of the vehicle and destination point are correctly oriented. At this time the vehicle enters an elliptical transfer to the stationary orbit height. At the apogee of this ellipse a maneuver occurs to simultaneously establish circular velocity and effect a plane change from the orbit plane to the equatorial plane.

Thus the maneuver consists of two propulsive phases separated by a coast phase. The first phase initiates the elliptical transfer while the second provides the circularization and a plane change. For these maneuvers thrust-to-weight ratios in the 0.1 to 0.5 range appear to be of interest.

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Planetary and Lunar Vicinity Maneuvers

These maneuvers originate in the vicinity of a body other than the earth. They include maneuvers initiated in an earth-moon/planet coast trajectory for the purpose of landing or orbiting and maneuvers initiated in the vicinity of the body for the purpose of establishing a planet/moon - earth return trajectory.

Direct Lunar Landing and Take-off. The direct lunar landing maneuver brings the vehicle to rest on the lunar surface directly from the earth-moon coast trajectory without making use of a coast period in a lunar orbit. Based upon energy requirements the method of accomplishing this maneuver consisted of two phases: (1) a thrust parallel to velocity (gravity turn) phase initiated in the earth-moon trajectory and (2) a relatively short vertical maneuver ending at the lunar surface. In studying this maneuver the trajectory was simulated by initiating thrust at the desired end point of the retro-thrust period (at the lunar surface) and flying backwards until the vehicle possesses the desired initial velocity of the earth-moon coast trajectory. From an energy consideration, desirable thrust-to-earth weight ratios for the main retro-thrust are in the 0.20 to 1.0 range (thrust-to-moon weight of 1.2 to 6.0). The velocity requirements for the direct landing are on the order of 8800 fps depending

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on the earth/moon transfer time. To achieve a desired moon-earth coast trajectory directly from the lunar surface a single propulsion period is required. This maneuver is very similar to the landing maneuver, however, energy requirements are slightly greater and the range of desirable thrust-to-weight ratios is more restrictive: 0.30 to 1.0. The velocity requirements are similar to those for landing.

Lunar Orbit Establishment and Departure. As a vehicle approaches the vicinity of the moon on its earth-moon trajectory it possesses a certain velocity which is dependent upon the earth-moon transfer time as discussed previously. To establish an orbit the vehicle must be decelerated to a given lunar circular orbit velocity. The maneuver used to accomplish this aligns vehicle thrust with vehicle velocity.

The energy requirements for lunar orbit establishment depend to a great extent upon the earth-moon trip time and the orbit height desired. For establishing a low altitude (50 n. mile) orbit thrust-to-earth-weight ratios in the 0.10 to 0.50 appear desirable based on consideration of the energy requirements. The velocity required for this maneuver based on a 2.6 day trip time are about 3240 fps. Departure from a lunar orbit to a moon-earth coast trajectory is essentially the reverse of the orbit establishment maneuver and was analyzed based on a thrust aligned with

[redacted]

velocity trajectory. The energy requirements depend upon the initial orbit height and the trip time desired. Thrust-to-weight ratios are similar to those in the orbit establishment maneuver.

<u>Landing from and Taking off to Lunar Orbit</u>. The requirement for an intermediate lunar parking orbit on either landing or take-off was studied similar to the above maneuvers by flying backwards from the lunar surface to the desired orbit. For this maneuver a trajectory utilizing an intermediate coast phase with a circularization phase at the desired orbit height was considered.

Results for this maneuver are similar to those of the direct landing or takeoff with the exceptions that the velocity requirements are lower and an engine restart is required. Desirable thrust-to-earth weight ratios are in the 0.4 to 1.0 range and a constant velocity descent phase would require throttling. The velocity requirement from a 50 n. mile orbit are about 5730 fps.

For the landing mission, a hovering or constant velocity descent phase several hundred feet above the lunar surface appears desirable for the initial lunar landing missions where detailed landing site surveillance has not been preconducted. Hovering will require approximately a thrust-to-(Earth) weight of 0.17 to provide the local lunar thrust-to-weight of 1.

For the moon mission, the total velocity increment for landing from an orbit 8970 ft/sec or direct 8850 ft/sec tends to indicate that single stage system could be employed. Thus, based on a stage which provides 10,000 ft/sec and has hydrogen/oxygen propellants with an initial thrust-to-(Earth) weight of 0.4, the burnout thrust ratio would be 0.835 (Earth), or 5.07 (lunar). Thus, to achieve a lunar thrust-to-weight of 1 for the hovering/landing phase, 5 to 1 propulsion throttling would be required.

Orbit Establishment Propulsion. Propulsion requirements for establishment of orbits about the Moon, Venus, and Mars have been investigated in conjunction with the Earth orbit propulsion phase. Nomographs for Venus and Mars orbit mission velocity requirements have been formulated.

Planetary orbit establishment and departure maneuvers are very similar to the corresponding lunar maneuvers. As a vehicle approaches a planet on its earth-planet trajectory, its velocity must be reduced to a given planet orbiting velocity. To accomplish this maneuver, the thrust is aligned opposing the velocity. The energy requirements depend upon the earth-planet trip time and the desired orbit height. Thrust-to-earth weight ratio effects are similar to those for the lunar maneuvers.

Based on the preliminary studies, thrust-to-(Earth) weight ratios for leaving or establishing an orbit about the Earth, moon, or planets can be arbitrarily selected within the limits indicated in the following table.

[redacted]

Earth	0.3 to 1.5
Moon	0.06 to 0.5
Venus	0.2 to 1.2
Mars	0.1 to 1.0

Within these limits propulsion maneuver velocity increments do not change greatly. There are, however, optimum values giving maximum payload.

Planetary Landing (Hovering) Requirements. Review of the requirements for landing or take-off maneuvers from Earth, moon, or planets, indicate more stringent limitations. For these latter maneuvers, the thrust-to-(Earth)weight ratios must be high enough to give a local acceleration greater than the local gravity of the body considered. For a hovering phase in the landing mission this thrust-to-weight ratio must be achieved at the end of the landing propulsion phase.

Earth	1.0
Moon	0.17
Venus	0.87
Mars	0.39

The approximate total velocity increments for landing from orbit (including the estimated losses) are 13,000 fps for Mars, and 25,000 fps for Venus. For both the Venus and Mars landing missions throttling would be required for the hovering maneuver.

[redacted]

[redacted]

Miscellaneous Maneuvers

These maneuvers are not necessarily tied to the vicinity of any planetary body and may be required at any number of positions in space.

Attitude Control. Attitude control requirements for a space vehicle can be defined in terms of the attitude angle and rate measured from an inertial reference system, and in the length of time used to achieve the required changes in these parameters. In general where the length of time is long and the required changes small, attitude control systems such as inertia wheels and electrical propulsion become attractive. For shorter times and more strenuous requirements, the jet-reaction devices are advantageous. For certain of the jet-reaction applications throttling appears attractive.

Methods were developed for the purpose of calculating torque, momentum, and propellant weight requirements for various angular attitude motions in a plane using jet-reaction devices. These parameters are expressed in ratio form to the moment of inertia of the space vehicle whose attitude is being controlled. The required angular maneuvers are expressed as a required change in attitude angle and rate during a given period of time. The control sequence is determined by assuming jet-reaction devices that may or may not be gimballed to produce the required changes in vehicle attitude.

[redacted]

<u>Mid-Course Corrections</u>. The mid-course correction maneuver is used to correct the coast trajectory for any errors introduced at burn-out of the initial propulsion phase of a space mission due to thrust variation, guidance error, tracking inaccuracy, etc. Information in the literature indicates that a single maneuver is sufficient to provide terminal position correction while use of two maneuvers provides both terminal position and velocity corrections for Lunar and planetary missions. From these considerations a single mid-course correction was assumed with the second (velocity correction) maneuver being combined with the gross maneuvers in the vicinity of the target body.

[redacted]

PRELIMINARY LIQUID PROPELLANT SPACE VEHICLE DESCRIPTION

For the array of booster systems and space missions considered, the space vehicles have been analyzed based on the propulsion systems described below (Table 1-5). A description of the assumed space missions is shown in Table 1-6.

Table 1-5

SPACE STAGE PROPULSION SYSTEM DESCRIPTION

Propellants	Feed System	Specific Impulse sec	Propellant Fraction
LO_2/LH_2	Pump	428	0.92
	Pressure	401	0.90
MON/MMH	Pressure	320	0.915
Nuclear	Pump	820	----

These space vehicle systems (Tables 1-7 to 1-16) should be considered as nominal, as only the major propulsion for these space stages was considered. The breakdown of the space vehicle stages indicated thrust level requirements for the various space maneuver missions. These preliminary recommended propulsion systems are based on the trajectory maneuver analysis conducted and the representative space vehicles which are reviewed.

[redacted]

			and simultaneously change plane,		
			110 day transfer, 28 January 1961	V_h = 9,800	Fig. 2-81
		(B)	Establish Venus Orbit	V_a = 16,400	Fig. 2-82
		(C)	Leave Venus Orbit and simultaneously change plane, 110 day transfer	V_h = 16,400	(Mirror Image)
		(D)	Earth orbit establishment	V_a = 9,800	(Return Trajectory) (Assumed)
7.	Soft Mars Landing	(A)	Depart from 300 n mi earth orbit and simultaneously change planes, 200 day transfer, 5 October 1960	V_h = 14,400	Fig. 2-79
		(B)	Establish Mars Orbit	V_a = 14,200	Fig. 2-80
		(C)	Land on Mars	V = 13,000	(Approximation)
8.	Mars Orbit and Return	(A)	Depart from 300 n mi earth orbit and simultaneously change planes, 200 day transfer, 5 October 1960	V_h = 14,400	Fig. 2-79
		(B)	Establish Mars Orbit	V_a = 14,200	Fig. 2-80
		(C)	Leave Mars orbit and simultaneously change planes, 200 day transfer	V_h = 14,200	(Mirror Image)
		(D)	Earth orbit establishment	V = 14,400	(Return Trajectory) (Assumed)
9.	24-Hour Orbit Establishment	(A)	Depart from 300 n mi orbit	V = 7,730	Fig. 2-27
		(B)	Plane change and circularize	V = 5,950	Fig. 2-29
10.	Orbital Rendezvous	(A)	Eccentricity change, 200 x 400 n mi to 300 x 400 n mi orbit	V = 170	Fig. 2-12
		(B)	Circularize and change planes	V = 2,220	Fig. 2-17

*These velocities were obtained from the interim report. They were used in some of the techniques developed for mission calculations in this report.

NOTE: V = ideal velocity increment
V_h = hyperbolic excess velocity
V_a = hyperbolic arrival velocity

TABLE 2-28 and 1-6

MISSION DESCRIPTION

Mission		Maneuver	Nominal Velocity* Requirement, fps	Report Reference
1. Soft Lunar Landing	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. 2-46
	(B)	Direct lunar landing	V = 8,920	Fig. 2-123
2. Lunar Satellite and	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. 2-46
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. 2-114, 2-112
	(C)	Depart from lunar orbit, 2.6 day earth transfer	V = 3,270	Fig. 2-115, 2-116
	(D)	Augmented Aerodynamic Re-entry	V = 5,000	
3. Soft Lunar Landing and Return to Earth's Surface	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. 2-46
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. 2-114, 2-112
	(C)	Land on moon	V = 5,730	Fig. 2-126
	(D)	Take-off to lunar orbit	V = 5,750	Fig. 2-127
	(E)	Depart from lunar orbit, 2.6 day earth transfer	V = 3,290	Fig. 2-115, 2-116
	(F)	Augmented Aerodynamic Re-entry	V = 5,000	
4. Soft Lunar Landing and	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. 2-46
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. 2-114, 2-112
	(C)	Land on moon	V = 5,730	Fig. 2-126
	(D)	Take off to lunar orbit	V = 5,750	Fig. 2-127
	(E)	Depart from lunar orbit, 2.6 day earth transfer	V = 3,290	Fig. 2-115, 2-116
	(F)	Establish earth orbit, 300 n mi	V = 10,180	Fig. 2-46
5. Soft Venus Landing	(A)	Depart from 300 n mi earth orbit and simultaneously change plane, [illegible] day transfer, 20 January 1961	V_h = [illegible]	Fig. 2-81
	(B)	Establish Venus Orbit	V_a = 14,400	Fig. 2-82
	(C)	Venus landing	V = 25,000	(Approximation)
6. Venus Orbit and Return	(A)	Depart from 300 n mi earth orbit		

TABLE [illegible]

MISSION DESCRIPTION

Mission		Maneuver	Nominal Velocity* Requirement, fps	Report Reference
1. Soft Lunar Landing	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. VIII D-7
	(B)	Direct lunar landing	V = 8,020	Fig. VIII [illegible]
2. Lunar Satellite and	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. VIII D-7
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. VIII G-15, 13
	(C)	Depart from lunar orbit 2.6 day earth transfer	V = 3,270	Fig. VIII G-16, 17
	(D)	Augmented Aerodynamic Re-entry	V = 5,000	
3. Soft Lunar Landing and Return to Earth's Surface	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. VIII D-7
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. VIII G-15, 13
	(C)	Land on moon	V = 5,730	Fig. VIII G-27
	(D)	Take-off to lunar orbit	V = 5,750	Fig. VIII G-28
	(E)	Depart from lunar orbit, 2.6 day earth transfer	V = 3,290	Fig. VIII G-16, 17
	(F)	Augmented Aerodynamic Recovery	V = 9,000	
4. Soft Lunar Landing and	(A)	Depart from 300 n mi earth orbit, 2.6 day lunar transfer	V = 10,180	Fig. VIII D-7
	(B)	Establish 50 n mi lunar orbit	V = 3,240	Fig. VIII G-15, 13
	(C)	Land on moon	V = 5,730	Fig. VIII G-27
	(D)	Take off to lunar orbit	V = 5,750	Fig. VIII G-28
	(E)	Depart from lunar orbit, 2.6 day earth transfer	V = 3,290	Fig. VIII G-16, 17
	(F)	Establish earth orbit, 300 n mi	V = 10,180	Fig. VIII D-7
5. Soft Venus Landing	(A)	Depart from 300 n mi earth orbit and simultaneously change plane, 115 day transfer 29 January 1962	V_h = 9,800	Fig. VIII E-32
	(B)	Establish Venus orbit	V_x = 16,500	Fig. VIII E-33

6. Venus Orbit and Return	(A)	Depart from 300 n mi earth orbit and simultaneously change plane, 110 day transfer, 28 January 1961	V_h = 9,800	Fig. VIII E-32
	(B)	Establish Venus Orbit	V_a = 16,400	Fig. VIII E-33
	(C)	Leave Venus Orbit and simultaneously change plane, 110 day transfer	V_h = 16,400	(Mirror Image Return Trajectory Assumed)
	(D)	Earth orbit establishment	V_a = 9,800	
7. Soft Mars Landing	(A)	Depart from 300 n mi earth orbit and simultaneously change planes, 200 day transfer. 5 October 1960	V_h = 14,400	Fig. VIII E-30
	(B)	Establish Mars orbit	V_a = 14,200	Fig. VIII E-31
	(C)	Land on Mars	V = 13,000	(Approximation)
8. Mars Orbit and Return	(A)	Depart from 300 n mi earth orbit and simultaneously change planes, 200 day transfer, 5 October 1960	V_h = 14,400	Fig. VIII E-30
	(B)	Establish Mars orbit	V_a = 14,200	Fig. VIII E-31
	(C)	Leave Mars orbit and simultaneously change planes, 200 day transfer	V_h = 14,200	(Mirror Image Return Trajectory Assumed)
	(D)	Earth orbit establishment	V_a = 14,400	
9. 24-Hour Orbit Establishment	(A)	Depart from 300 n mi orbit	V = 7,730	Fig. VIII C-6
	(B)	Plane change and circularize	V = 5,950	Fig. VIII C-8
10. Orbital Rendezvous	(A)	Eccentricity change, 200 x 400 n mi to 300 x 400 n mi orbit	V = 170	Fig. VIII B-4
	(B)	Circularize and change planes	V = 2,220	Fig. VIII B-9

*These velocities were obtained from the interim report. They were used in some of the techniques developed for mission calculations in this report.

NOTE: V = ideal velocity increment
V_h = hyperbolic excess velocity
V_a = hyperbolic arrival velocity

[redacted]

A

TABLE

SPACE PROPULSI

MISSION: Soft, Direct

PROPULSION SYSTEM:

BOOSTER VEHICLE	SPACE STAGE ONE GROSS WEIGHT (LB)	SPACE STAGE ONE THRUST (LB)	
Atlas Centaur	8,500	4,250	3A
Saturn C-1	24,300	12,100	1B
Saturn C-2	50,630	25,300	3B
Saturn: Nuclear	---	---	
Nova: R-2	73,000	36,400	4B
Nova: H-2	118,000	59,000	6B
Nova: R-4	146,000	73,000	1C
Nova: H-4	236,000	118,000	2C
Nova: H-6	354,000	177,000	2C
Nova: N-4	---	---	

BASIC PROPULSION UNITS:

A - 1.5K
B - 10 K
C - 75 K
} High Energy

D - 150 lbs
E - 1.0K
F - 7.5K
} Storable

ST

B

[redacted]

1-7

ON SYSTEM DESCRIPTION

Lunar Landing (No Return,

LO_2/LH_2 Pump-Fed

SPACE STAGE TWO GROSS WEIGHT (LB)	THRUST (LB)		PAYLOAD LB
3,600	2,500	2A	1,750
10,500	7,300	5A	5,000
21,400	15,000	1B	10,300
31,700	22,000	2B	15,200
31,000	22,000	2B	15,000
50,000	35,000	4B	24,000
62,000	44,000	4B	30,000
100,000	70,000	1C	48,000
150,000	100,000	1C	72,000
261,000	180,000	2C	126,000

STAGE	FUNCTION
1	Leave Earth orbit
2	Lunar Landing

[redacted]

A

SPACE PROPU

MISSION: Lunar Satellite an

PROPULSION S

BOOSTER VEHICLE	SPACE STAGE ONE GROSS WEIGHT (LB)	SPACE STAGE ONE THRUST (LB)	
Atlas Centaur	8,500	4,250	3A
Saturn C-1	24,300	12,100	1B
Saturn C-2	50,630	25,300	3B
Saturn: Nuclear	---	---	
Nova: R-2	73,000	36,400	4B
Nova: H-2	118,000	59,000	6B
Nova: R-4	146,000	73,000	1C
Nova: H-4	236,000	118,000	2C
Nova: H-6	354,000	177,000	2C
Nova: N-4	---	---	

BASIC PROPULSION UNITS:

A - 1.5K
B - 10K
C - 75K
} High Energy

D - 150 lbs
E - 1.0K
F - 7.5K
} Storable

STAG
1

2

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ABLE 1-8

LSION SYSTEM DESCRIPTION

Return to Earth Surface (Aerodynamically)

YSTEM: LO_2/LH_2 Pump-fed

SPACE STAGE TWO GROSS WEIGHT (LB)	SPACE STAGE TWO THRUST (LB)		PAYLOAD LB
2,700	1,300	1A	1,370
7,500	3,500	2A	3,900
16,000	8,000	5A	8,100
31,700	16,000	2B	12,400
23,000	11,500	1B	11,700
37,000	18,500	2B	18,500
46,000	23,000	2B	23,400
75,000	37,500	4B	38,100
111,000	56,000	6B	56,500
261,000	130,000	2C	101,000

E

FUNCTION

Leave Earth orbit
Establish Lunar orbit

Leave Lunar orbit
Augment Aerodynamic landing

A1

SPACE PRO[illegible]

MISSION: Soft Lunar Landing [illegible]

PROPULSION SYSTEM: Stages 1 & 2 - LO_2/LH_2, Pump-Fed[illegible]

[illegible] VEHICLE	STAGE ONE GROSS WEIGHT (LB)	STAGE ONE THRUST (LB)		ST[illegible] GROSS WEIGHT (LB)	T[illegible]
[illegible] Centaur	8,500	4,250	3[illegible]	3,600	1[illegible]
[illegible] C-1	24,300	12,100	1B	10,500	4[illegible]
[illegible]	50,630	25,300	3B	21,400	[illegible]
[illegible]	—	—		35,170	1[illegible]
Nova: [illegible]-2	73,000	36,400	4B	31,000	12,[illegible]
Nova: E-2	118,000	59,000	6B	50,000	20,[illegible]
Nova: [illegible]-4	146,000	73,000	1C	62,000	24,[illegible]
Nova: [illegible]-4	236,000	118,000	2C	100,000	40,[illegible]
Nova: [illegible]-6	354,000	177,000	3C	150,000	60,[illegible]
Nova: N-4	—	—		261,000	100,[illegible]

* The two payload reflect the effects of using LO_2/LH_2 or a storable propellant combination ([illegible]/[illegible]) in the third stage.

BASIC [illegible] UNITS:

A - 1.5K
B - [illegible]K
C - [illegible]K } High Energy

D - 150 lbs
E - 1.0K
F - 7.5K } Storable

[redacted]

AZ

SION SYSTEM DESCRIPTION

d Return to Earth Surface (Aerodynamically)

Stage 3 - LO_2/LH_2 or Storable, Pressure-Fed as Indicated

SPACE STAGE TWO THRUST (LB)		SPACE STAGE THREE GROSS WEIGHT (LB)	SPACE STAGE THREE THRUST (LB)		PAYLOAD* ~ LB LO_2/LH_2	PAYLOAD* ~ LB STORABLE
,400	1A	1,000	200	2D	450	400
,000	3A	2,900	600	4D	1,350	1,150
,500	[illegible]	6,000	1,200	1E	2,800	2,400
,500	1B	9,000	1,800	2E	4,160	3,570
.000	1B	8,700	1,750	2E	4,100	3,500
000	2B	14,100	2,800	3E	6,700	5,600
500	3B	17,500	3,500	4E	8,300	7,000
000	4B	28,200	5,700	6E	13,300	11,300
000	6B	42,300	8,500	1F	19,900	16,900
000	2C	73,700	14,500	2F	35,000	29,200

STAGE	FUNCTION	STAGE	FUNCTION
1	Leave Earth orbit	3	Leave Lunar orbit Augment Aerodynamic landing
2	Establish Lunar orbit Land on Moon Lunar launch to orbit		

THIS MATERIAL CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES [illegible] IS PROHIBITED BY LAW

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B1

SPACE PRO

MISSION: Soft Lun

PROPULSION SYSTEM: Stages 1 & 2 - LO_2/LH_2, Pump-Fed

BOOSTER VEHICLE	SPACE STAGE ONE GROSS WEIGHT (LB)	THRUST (LB)		GROSS WEIGHT (LB)
Atlas Centaur	8,500	4,250	3A	3,600
Saturn C-1	24,300	12,100	1B	10,500
Saturn C-2	50,630	25,300	3B	21,400
Saturn: Nuclear	---	---	---	33,170
Nova: R-2	73,000	36,400	4B	31,000
Nova: H-2	118,000	59,000	6B	50,000
Nova: R-4	146,000	73,000	1C	62,000
Nova: H-4	236,000	118,000	2C	100,000
Nova: H-6	354,000	177,000	2C	150,000
Nova: N-4	---	---	---	261,000

* The two payloads reflect the effects of using LO_2/LH_2 or a storable propellant com
(MON/MMH) in the third stage.

BASIC PROPULSION UNITS:

A - 1.5K
B - 10K
C - 75K } High Energy

D - 150 lbs
E - 1.0K
F - 7.5K } Storable

82

[redacted]

TABLE 1-10

PULSION SYSTEM DESCRIPTION

ar Landing and Return to Earth Orbit

; Stage 3 - LO_2/H_2 or Storable; Pressure-Fed as Indicated

SPACE STAGE TWO THRUST (LB)		SPACE STAGE THREE GROSS WEIGHT (LB)	SPACE STAGE THREE THRUST (LB)		PAYLOAD* LB LO_2/LH_2	PAYLOAD* LB STORABLE
1,400	1A	1,000	200	2D	250	200
4,000	3A	2,900	600	4D	800	600
8,500	6A	6,000	1,200	1E	1,650	1,250
12,500	1B	9,000	1,800	2E	2,450	1,850
12,500	1B	8,700	1,750	2E	2,350	1,800
20,000	2B	14,100	2,800	3E	3,850	2,900
24,500	3B	17,500	3,500	4E	4,750	3,600
40,000	4B	28,200	5,700	6E	7,650	5,850
60,000	6B	42,300	8,500	1F	11,500	8,750
100,000	2C	74,000	14,500	2F	20,000	15,200

bination

STAGE	FUNCTION
1	Departure from earth orbit
2	Establish Lunar orbit Land on moon Lunar launch to orbit
3	Depart Lunar orbit Establish Earth orbit

[redacted]

01

TABLE I-11

SPACE PROPULSION SYSTEM DE

MISSION: VENUS LANDING (NON-AE

PROPULSION SYSTEM: LO_2/LH

VEHICLE	SPACE STAGE ONE GROSS WEIGHT(lb)	THRUST(lb)		SPACE GROSS WEIGH
Atlas Centaur	8,500	4,250	3A	3,258 Fr
Saturn C-1	24,300	12,100	1B	9,320
Saturn C-2	50,600	25,300	3B	19,400
Saturn Nuclear	---	---		34,600
Nova R-2	73,000	36,400	4B	27,900
Nova H-2	118,000	59,000	6B	45,200
Nova R-4	146,000	73,000	1C	55,900
Nova H-4	236,000	118,000	2C	90,500
Nova R-6	354,000	177,000	2C	136,000
Nova N-4	---	---		270,800

BASIC PROPULSION UNITS:

A - 1.5K } High Energy
B - 10K
C - 25K
D - 150 lbs } Storable
E - 1.0K
F - 2.5K

OZ

SCRIPTION

RODYNAMIC)

2 PUMP-FED

STAGE TWO			SPACE STAGE THREE			
T(lb)	THRUST(lb)		GROSS WEIGHT(lb)	THRUST(lb)		PAYLOAD(lbs)
obe	1,700	1A	1,263	1,520	1A	267
	4,700	3A	3,615	[illegible]	[illegible]	[illegible]
	10,000	1B	7,5[illegible]	[illegible],900	[illegible]	[illegible]
	[illegible],000	2B	13,4[illegible]	[illegible]		[illegible]
	14,000	1B	10,[illegible]	[illegible]		[illegible]
	23,000	2B	17,[illegible]50	2[illegible]		[illegible]
	28,000	3B	21,700	2[illegible],000	[illegible]	[illegible]
	45,000	5B	35,[illegible]	[illegible]	[illegible]	[illegible]
	62,000	1C	52,[illegible]	63,[illegible]		[illegible]
	1[illegible],000	2C	105,000	[illegible]		[illegible]

[illegible] [illegible]
Desert [illegible]

P1

TABLE 1-12

SPACE PROPULSION SYSTE

MISSION: VENUS ORBIT AND RET

PROPULSION SYSTEM: STAGES 1 &

STAGES 3 &

VEHICLE	SPACE STAGE ONE GROSS WT. (lb)	SPACE STAGE ONE THRUST (lb)		SPACE STAGE TWO GROSS WT. (lb)	SPACE STAGE TWO THRUST (lb)		SPA GROSS W
Atlas Centaur	8,500	4,250	3A	3,258	1,600	1A	1,26
Saturn C-1	24,300	12,100	1B	9,320	4,700	3A	3,61
Saturn C-2	50,630	25,300	3B	19,400	9,700	1B	7,54
Saturn Nuclear	—	—		34,600	17,000	2B	13,44
Nova R-2	73,000	36,400	4B	27,900	15,000	2B	10,85
Nova B-2	118,000	59,000	6B	45,200	23,000	2B	17,55
Nova R-4	146,000	73,000	1C	55,900	28,000	3B	21,70
Nova N-4	236,000	118,000	2C	90,500	45,000	5B	35,10
Nova H-6	354,000	177,000	2C	136,000	68,000	1C	52,60
Nova N-4	—	—		270,800	140,000	2C	105,00

BASIC PROPULSION UNITS:

A - 1.5K }
B - 10K } High Energy
C - 75K }
D - 150 lbs }
E - 1.0K } Storable
F - 7.5K }

D2

[redacted]

M DESCRIPTION

JRN TO EARTH ORBIT

2 - LO_2/LH_2 Pump-Fed

4 - MON/MMH Pump-Fed

CE STAGE THREE			SPACE STAGE FOUR				
T.(lb)	THRUST(lb)		GROSS WT.(lb)	THRUST(lb)		PAYLOAD(lb)	PAYLOAD* (lb)
3	630	4D	321	160	1D	84.8	176
5	1,800	2E	918	450	3D	242	504
0	3,700	4E	1,910	960	1E	505	1,050
0	6,700	1F	3,415	1,800	2E	901	1,870
0	5,400	5E	2,760	1,400	1E	728	1,510
0	8,800	1F	4,450	2,200	2E	1,177	2,440
0	11,000	2F	5,520	2,800	3E	1,456	3,020
0	18,000	2F	8,900	4,500	5E	2,354	4,880
0	26,000	3F	13,380	6,600	1F	3,530	7,340
0	50,000	6F	26,700	14,000	2F	7,050	14,650

* Payload for vehicles using a LO_2/LH_2 Pump-fed propulsion system for all four stages.

STAGE	FUNCTION
1	Depart from earth orbit
2	Establish orbit about Venus
3	Depart Venus orbit
4	Establish Earth orbit

[redacted]

E1

TABLE 1-

SPACE PROPULSION

MISSION: Mars

PROPULSION SYSTEM:

BOOSTER VEHICLE	SPACE STAGE ONE GROSS WEIGHT (lb)	SPACE STAGE ONE THRUST (lb)		SPAC[illegible] GROSS WEIGHT (lb)
Atlas Centaur	8,500	4,250	3A	2,770 F
Saturn C-1	24,300	12,100	1B	7,900
Saturn C-2	50,630	25,300	3B	16,500
Saturn Nuclear	—	—		23,340
Nova R-2	73,000	36,400	4B	23,600
Nova H-2	118,000	59,000	6B	38,500
Nova R-4	146,000	73,000	1C	47,500
Nova H-4	236,000	118,000	2C	76,900
Nova H-6	354,000	177,000	2C	115,000
Nova N-4	—	—		193,300

BASIC PROPULSION UNITS:

A -1.5K
B - 10K
C - 75K
D -1[illegible] lbs
E -3.0K
F -7.5K

A, B, C: High Energy

D, E, F: Storable

EZ

[redacted]

YSTEM DESCRIPTION

Landing (Non-Aerodynamic)

O_2/LH_2 Pump-Fed

STAGE TWO		SPACE STAGE THREE			PAYLOAD
THRUST (lb)		GROSS WEIGHT (lb)	THRUST (lb)		(lb)
be 550	1A	1,280	7[illegible]0	1A	469
1,580	1A	3,640	2,180	2A	1,330
2,300	2A	7,600	4,550	3A	2,[illegible]
4,660	3A	10,730	6,450	4A	3,940
4,700	3A	10,940	6,550	4A	4,000
7,700	5A	17,700	10,600	1B	6,490
9,500	1B	21,900	13,1[illegible]	1B	8,000
15,400	2B	35,400	21,[illegible]	2B	13,[illegible]
23,000	3B	53,000	31,[illegible]	3B	19,400
38,600	4B	89,000	[illegible]	5B	32,600

FUNCTION
Leave the earth orbit
Establish Mars orbit
Land from orbit

[redacted]

F1

TABLE 1

SPACE PROPULSION ...

MISSION: MARS ORBIT AND ...

PROPULSION SYSTEM: STAGES 1 ...

STAGES 3

BOOSTER VEHICLE	SPACE STAGE ONE GROSS WEIGHT (LB)	SPACE STAGE ONE THRUST (LB)		SPACE STAGE TWO GROSS WEIGHT (LB)	SPACE STAGE TWO THRUST (LB)
Atlas Centaur	8,500	4,250	3A	2,770	550
Saturn C-1	24,300	12,100	1B	7,900	1,580
Saturn C-2	50,630	25,300	3B	16,500	3,300
Saturn Nuclear	—	—		23,340	4,660
Nova R-2	73,000	36,400	4B	23,600	4,700
Nova S-2	118,000	59,000	6B	38,500	7,700
Nova R-4	146,000	73,000	1C	47,500	9,500
Nova S-4	236,000	118,000	2C	75,900	15,400
Nova S-6	354,000	177,000	2C	115,000	23,000
Nova N-4	—	—		193,300	38,600

BASIC PROPULSION UNITS:

A - 1.5K
B - 10K
C - 75K
} High Energy

D - 150 K...
E - 1.0K
F - 7.5K
} Storable

F2

[redacted]

M DESCRIPTION

URN TO EARTH ORBIT

2 - LO_2/LH_2 PUMP-FED

4 - MON/MMH PUMP-FED

	SPACE STAGE THREE GROSS WEIGHT (LB)	SPACE STAGE THREE THRUST (LB)		SPACE STAGE FOUR GROSS WEIGHT (LB)	SPACE STAGE FOUR THRUST (LB)		PAYLOAD (LB)	PAYLOAD* (LB)
	1,280	256	2D	440	220	2D	110	190
A	3,640	730	1E	1,270	640	4D	310	540
A	7,600	1,520	2E	2,640	1,320	1E	645	1,120
A	10,730	2,140	2E	3,730	1,870	2E	910	1,580
A	10,940	2,180	2E	3,800	1,950	2E	930	1,610
A	17,700	3,540	4E	6,150	3,000	3E	1,500	2,600
B	21,900	4,350	4E	7,600	3,800	4E	1,860	3,220
B	35,400	7,100	1F	12,300	6,150	6E	3,000	5,200
B	53,000	10,600	2F	18,400	9,200	1F	4,500	7,800
B	89,000	17,800	2F	31,000	15,500	2F	7,550	13,400

* Payload for vehicles using a LO_2/LH_2 Pump-Fed propulsion system for all four stages.

STAGE	FUNCTION
1	Leave the earth orbit
2	Establish Mars orbit
3	Depart Mars orbit
4	Establish Earth orbit

R-3208

[redacted]

[redacted]

TABLE 1-15

SPACE PROPULSION SYSTEM DESCRIPTION

MISSION: 24 HOUR CIRCULAR ORBIT

PROPULSION SYSTEM: LO_2/LH_2

SPACE STAGE

VEHICLE	GROSS WEIGHT (LB)	THRUST (LB)		PAYLOAD (LB)
Atlas-Centaur	8,500	4,250	3A	2,600
Saturn C-1	24,300	12,150	1B	7,500
Saturn C-2	50,600	25,300	3B	15,600
Saturn Nuclear	—	—		24,300
Nova R-2	73,000	36,500	4B	22,500
Nova H-2	108,000	54,000	6B	33,300
Nova R-4	146,000	73,000	1C	45,000
Nova H-4	236,000	118,000	2C	72,700
Nova H-6	354,000	177,000	2C	109,000
Nova N-4	—	—		198,800

BASIC PROPULSION UNITS:

$$\left.\begin{array}{l} \text{A} - 1.5\text{K} \\ \text{B} - 10\text{ K} \\ \text{C} - 75\text{K} \end{array}\right\} \text{High Energy}$$

$$\left.\begin{array}{l} \text{D} - 150\text{ lbs} \\ \text{E} - 1.0\text{K} \\ \text{F} - 7.5\text{K} \end{array}\right\} \text{Storable}$$

[redacted]

TABLE 1-16

SPACE STAGE PROPULSION SYSTEM DESCRIPTION MISSION

300-n mi Rendezvous From 5 deg Inclined 300/200-n mi Ellipse

Booster Vehicle, lb	Space Stage		Payload lb
	Gross Weight, lb	Thrust, lb	
Atlas-Centaur	8,500	850	7,000
Saturn C-1	24,500	2,450	[illegible]
Saturn C-2	50,650	5,065	[illegible]
Saturn Nuclear	68,000	6,800	[illegible]
Nova R-2	75,000	7,500	[illegible]
Nova H-2	118,000	11,800	[illegible]
Nova R-3	146,000	14,600	[illegible]
Nova H-3	256,000	25,600	[illegible]
Nova [illegible]-6	354,000	35,400	[illegible]
Nova N-3	505,000	[illegible]	[illegible]

Examination of the thrust levels resulting (for the space stages) suggests three possible approaches in designing space propulsion systems for these missions.

The most obvious approach and yet the costliest in time and money would be a propulsion system for each of the indicated thrust levels. This presupposes that all 10 vehicles would be used for all missions. A multitude of different engines would be necessary to supply the propulsion for all of the vehicle stages. A payload review indicates some of the vehicles would serve no important use for many of the missions selected. Those vehicles that appear to be most likely for the mission were noted (shaded areas). Thus, logically only propulsion systems for these particular vehicles would be required.

The second approach is derived from consideration of the propulsion requirements. The space vehicle selection for the missions indicates two possible propulsion needs for space missions. Because of the storage problem in the deep-space missions, both high-energy propellant and storable propellant systems may be employed. From the thrust requirements in the charts, thrust levels can be selected such that a limited number of high-energy propellant propulsion systems and

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storable propellant propulsion systems can satisfy the demands of any of the 10 vehicles listed. Three high-energy propellant systems can be selected, one each in the following ranges of thrust:

1. 1500 to 5000 lb
2. 10,000 to 25,000 lb
3. 75,000 to 100,000 lb

For storables, the thrusts can be arranged into ranges of:

1. 150 to 500 lb
2. 1000 to 5000 lb
3. 7500 to 20,000 lb

If the higher thrust of these ranges is chosen for basic propulsion system units, each system is applicable to several vehicles in one or more stages.

In applying one of these basic propulsion units to a vehicle, the thrust-to-weight ratio must be maintained within the limits indicated previously. Nonoptimum thrust-to-weight ratios are obtained, and the payload loss must be considered. For instance, a 25,000 lb basic propulsion unit may be applied as a stage for one vehicle and its initial thrust-to-weight would be as low as 0.2 for a particular propulsion phase. In application to another vehicle, the thrust-to-weight may be as large as 1.2 (Table 1-17). This possible performance

[redacted]

loss must be analyzed to ascertain the exact penalty for such an approach to propulsion requirements for space missions. Thus, more than the indicated number of basic units may be required to minimize performance loss. However, from the preliminary analysis, it appears feasible to use only the basic units.

The third approach of designing space engine systems (Table 1-18) would be clustering of basic units to accomplish thrust levels near the optimum thrust-to-weight ratios for the propulsive maneuver. For the high-energy propellant systems, selecting the lower limit thrust in each of the three ranges would give basic units of 1500, 10,000, and 75,000 lb thrust. For storable propellants, thrust levels of 150, 1000, and 7500 lb can be selected.

[redacted]

TABLE 1-17

APPLICATION OF SINGLE UNIT ENGINES

Range of Stage Gross Weights, lb	Unit Engine Thrust, 1000 lb	Range of Thrust-to-Earth Weight Ratio
3,600 15,000	5	1.40 0.333
15,000 100,000	25	1.67 0.25
100,000 354,000	100	1.00 0.28

TABLE 1-18

THRUST ACHIEVABLE WITH CLUSTERED UNIT ENGINES

Unit Engine Thrust, 1000 lb	Units in Cluster					
	1	2	3	4	5	6
1.5	1.5	3	4.5	6.0	7.5	9.0
10	10	20	30	40	50	60
75	75	150	225	300	375	—

[redacted]

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These basic units chambers, feed systems, and tanks can be used singly or clustered to give a wide choice of thrust levels. In clustering, the system inert weight may be larger than for a single unit of the same thrust level; however, this weight penalty may be less important than the simplicity, reliability, and development cost associated with designing propulsion systems for each of the thrust levels needed for the many missions analyzed. It may also be less than the loss in payload performance when using a thrust-to-weight ratio too far from optimum.

The number and type of basic units applicable to each vehicle and mission is indicated on the charts in the section entitled General Vehicle/Mission Description. Although only particular vehicles and missions will be recommended for further study, it is clear that selection of these basic propulsion systems for design ensures propulsion systems applicable to any of the vehicles and missions.

Secondary constraints imposed upon the design of propulsion systems, especially for some of the propulsion maneuvers such as landing, may require engine throttleability. Other things such as propellant storage, propellant volume, and redundancy of engines enter into system selection and may obviate some of the approaches described above. Therefore, when subjected to the further constraints, one

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of the approaches may appear as the only solution to the problems. Preliminary analysis, however, indicates that any of the three approaches can be taken to provide the propulsion necessary to accomplish any of the missions presented herein with the selected vehicles.

VEHICLE/MISSION COMBINATIONS RECOMMENDED FOR FURTHER STUDY

In the Phase 1 studies the numerous, broad aspects of space mission propulsion requirements have been considered. For space missions of interest propulsive maneuvers and propulsion systems were combined to describe space vehicles (initiated in a 300 n mi orbit) to accomplish these missions based upon current and future booster vehicles. It was concluded that in these vehicles the Earth (vicinity) initiated stages would use high-energy, cryogenic propellants. For space stages initiated after extremely long coast periods storable propellants may be advantageous; a detailed design comparison of the two types of propellants should be made considering insulation requirements and based on a detailed stage layout.

Various methods of selecting engines for use in the different vehicles were considered and either (1) clustering basic engine units to achieve new optimum thrust-to-weight ratios or (2) using single

[redacted]

basic engine units to operate at possible nonoptimum thrust levels appeared to have advantages depending upon their application.

From the array of missions/vehicles considered certain combinations were singled out as worthy of more detailed study. These mission/vehicle selections are made to reduce the propulsion systems to be studied in detail; in this case a system which typifies the combinations would be considered. Selections can also be made of individual combinations which are of particular interest. Following these two lines of thought 10 vehicle/mission combinations were suggested for more detailed study, (Table 1-19). These combinations cover the gamut of size and maneuver requirements. Placing primary emphasis upon five (checked systems) of these combinations and considering the remainder in a somewhat secondary manner will serve to facilitate the detailed study.

Propulsion systems can be optimized and designed for these particular vehicles. By determining and remaining aware of the possibilities and penalties of off-design operation, these propulsion systems can be efficiently utilized for vehicles other than those considered.

[redacted]

TABLE 1-19

VEHICLE/MISSION COMBINATION SELECTIONS*

Mission	Booster Vehicle	Stages	Nominal Thrust, 1000, lb	Nominal Payload, 1000 lb	Comments
24 Hour Orbit	Saturn C-1	1	12.2	7.5	Restart
Soft, Direct Lunar Landing	Atlas Centaur	1 2	4.5 3	1.75	Throttling
Lunar Satellite & Return (Aero)	Saturn C-2	1 2	30 7.5	8.[illegible]	Restart
Soft Lunar Land & Return (Aero)	Saturn C-2	1 2 3	30 [illegible] 1	[illegible]	Throttling & Restart
Soft Lunar Land & Return (Aero)	Nova N-2	1 2 3	70 20 [illegible]	[illegible].6	Throttling & Restart Restart
Soft Lunar Land & Return (Aero)	Nova N-[illegible]	1 2 [illegible]	[illegible] [illegible] [illegible]	[illegible]	Throttling & Restart Restart
Mars Landing (Soft)	Nova N-[illegible]	1 2 3	150 20 20	[illegible]	
Mars Probe	Atlas Centaur				
Mars Orbit & Return	Nova N-[illegible]	1 2 3 4	150 30 [illegible] [illegible]	[illegible]	
Orbital Rendezvous	Saturn C-2	1	[illegible]	[illegible]	

*The vehicle/mission combinations [illegible] were selected by NASA for consideration in the Phase [illegible] effort.

[redacted]

DETAILED PROPULSION SYSTEM INVESTIGATION

The Phase 2 investigation was conducted to analyze and describe the requirements of the three space missions Table 1-20 that were recommended for further study in Phase 1. Necessary to the analysis of all three missions is the study of certain propulsion system features. These features and their effects on the propulsion system can be studied to a great extent independent of a particular space mission.

Space Environment

The various constituents of the space environment were found to have significant effects on the space propulsion system. The conditions of hard vacuum, particulate radiation, zero gravity, meteoroids, and heat transfer are all such that the operation of a propulsion system in space would be seriously compromised unless the proper design procedures are followed.

Hard vacuum and particulate radiation present largely a material selection problem. By designing the propulsion system with materials which do not sublimate in vacuum or deteriorate under particulate radiation these problems can be circumvented. Tables 3-5 and 3-7 provide a guide to material selection for space applications.

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[redacted]

TABLE 1-20

SELECTED MISSION/VEHICLE COMBINATIONS

Space Mission	Booster Vehicle	Nominal Earth Escape Payload, lb
Soft, Lunar Landing and Return to Earth's Surface (Aero)	Nova H-6	150,000 (Vary from Nova H-8 to Saturn C-2)
Mars Orbit (No Return)	Nova H-6	150,000 (Vary from Nova H-8 to Saturn C-2)
Orbital Rendezvous	Nova H-2*	- - - - - - -

*Substituted for C-2

[redacted]

[redacted]

Zero gravity and meteoroids present design problems that are receiving considerable attention. Difficulty of separation of gas and liquid in propellant tanks is one of the more significant problems brought about by zero gravity conditions. Numerous methods of accomplishing this have been suggested. Many of these are feasible and can be incorporated in propulsion system design. Protection of propulsion systems from puncture by meteoroids appears to be provided most efficiently by "Whipple meteoroid bumpers." These thin shields surrounding the component to be protected seem to be very effective in reducing the penetration of the high energy meteoroids: Additional effort is necessary to provide good design information.

Heat transfer in space presents problems in storing propellants for extended times. These problems arise not only from the thermal radiation emitted by the sun and planet, but from conductive heat transfer which occurs between dissimilar propellants or between the propellants and other internal heat sources. Studies indicate that the "earth storable" propellants (hydrazine, etc.) can be easily maintained through proper surface and attitude control. For the missions currently contemplated the cryogenic propellants (hydrogen, etc.) can be maintained by surface and attitude control in combination with the application of a good insulation material such as Linde SI-4.

[redacted]

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Missions of long duration may require so much insulation for the cryogenic (high energy) propellant combinations that a storable propellant system will be able to provide more payload capability. This is illustrated by Fig. 1-3 developed in this study. The figure presents the combination of storage time and propellant weight which causes the payload of the cryogenic (LO_2/LH_2) combination to decrease to that of the storable propellant (MON/MMH) combination. If the combination of storage time and propellant weight results in a point above the curve the MON/MMH combination will provide the greater payload, if below the curve the LO_2/LH_2 combination will provide the greater payload.

Studies similar to that above indicated that for the present space vehicle, based on the NOVA H-6 booster, LO_2/LH_2 could be used in all stages for both lunar and Mars missions. The storability of these propellants is strongly dependent upon the internal conduction, the size of the vehicle in question, and the method of storage (no loss vs propellant boiloff). These require additional study before a complete evaluation can be made. The internal conduction in particular is a function of the detailed design of the vehicle and is difficult to analyze in a general manner.

Heat conduction between propellant tanks = 10 BTU/hr.
No storage losses.
Pump-fed systems.
Linde SI-4 insulation.

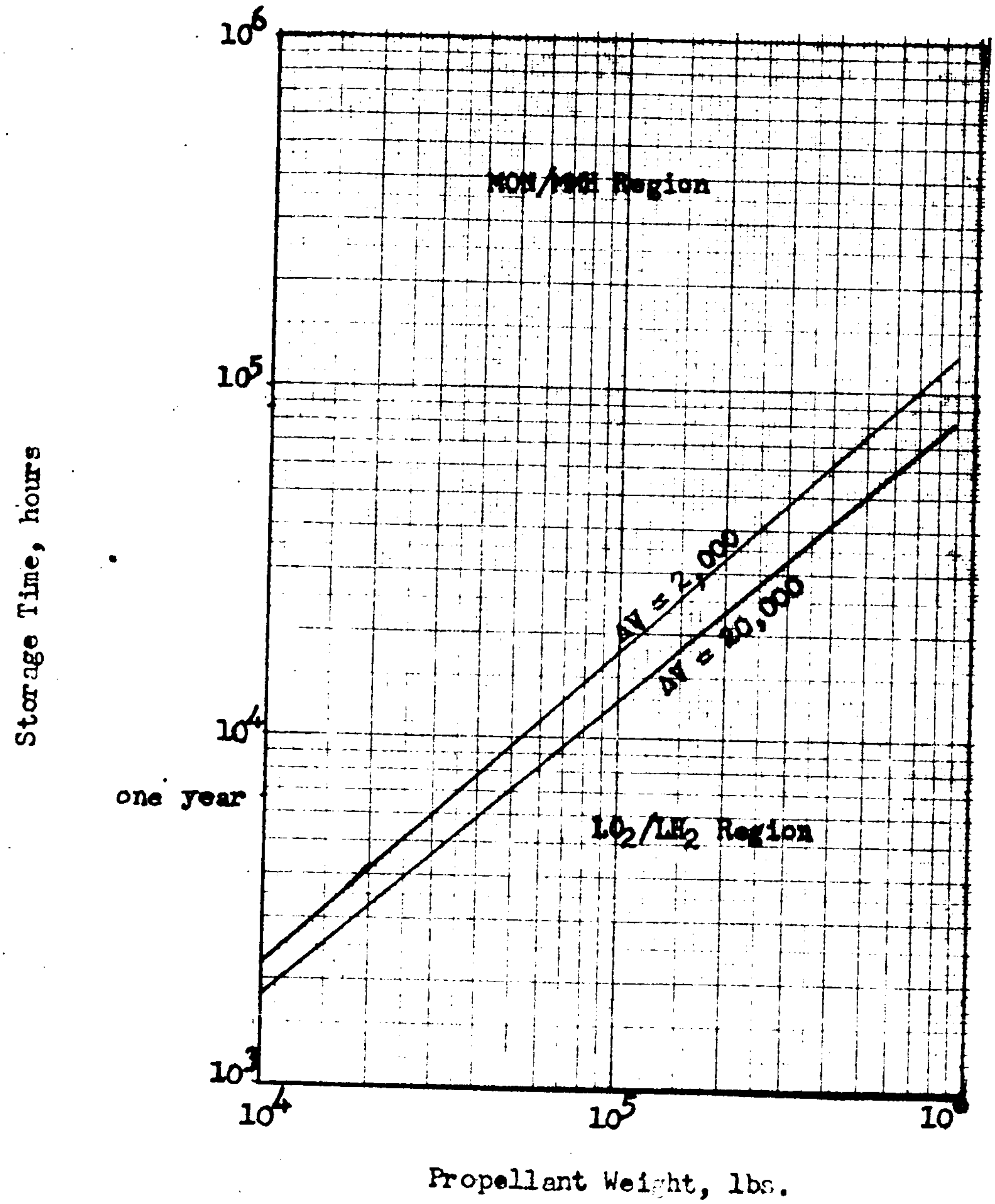

Figure 1-3. Effect of Space Storage Time on Propellant Selection

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ENGINE PARAMETER OPTIMIZATION

It is generally desirable to utilize the propulsion that provides the maximum payload capability for a given gross weight. This payload capability is strongly a function of the engine operating parameters: mixture ratio, thrust-to-weight ratio, chamber pressure, and expansion ratio. By proper selection of these parameters maximum payload capability can be provided. Through consideration of previous Rocketdyne studies parameters were selected for the preliminary propulsion systems to be used in the various mission studies.

Methods are also developed for the rapid selection of these operating parameters given certain hardware information. Using these methods optimum chamber pressure, expansion ratio, and thrust-to-weight ratio can be determined. The effect of various factors on the optimum chamber pressure is illustrated in Figure 1-4 for pump and pressure-fed systems. Variation of $\pm$ 50% in certain factors (thrust-to-weight ratio, η ; pump weight factor, ν ; thrust chamber weight factor, ψ ; bulk density, ρ_B ; tank weight factor, γ ; pressure drop factor, $\frac{\partial P_D}{\partial P}$) affect the optimum chamber pressure as shown.

[redacted]

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[redacted]

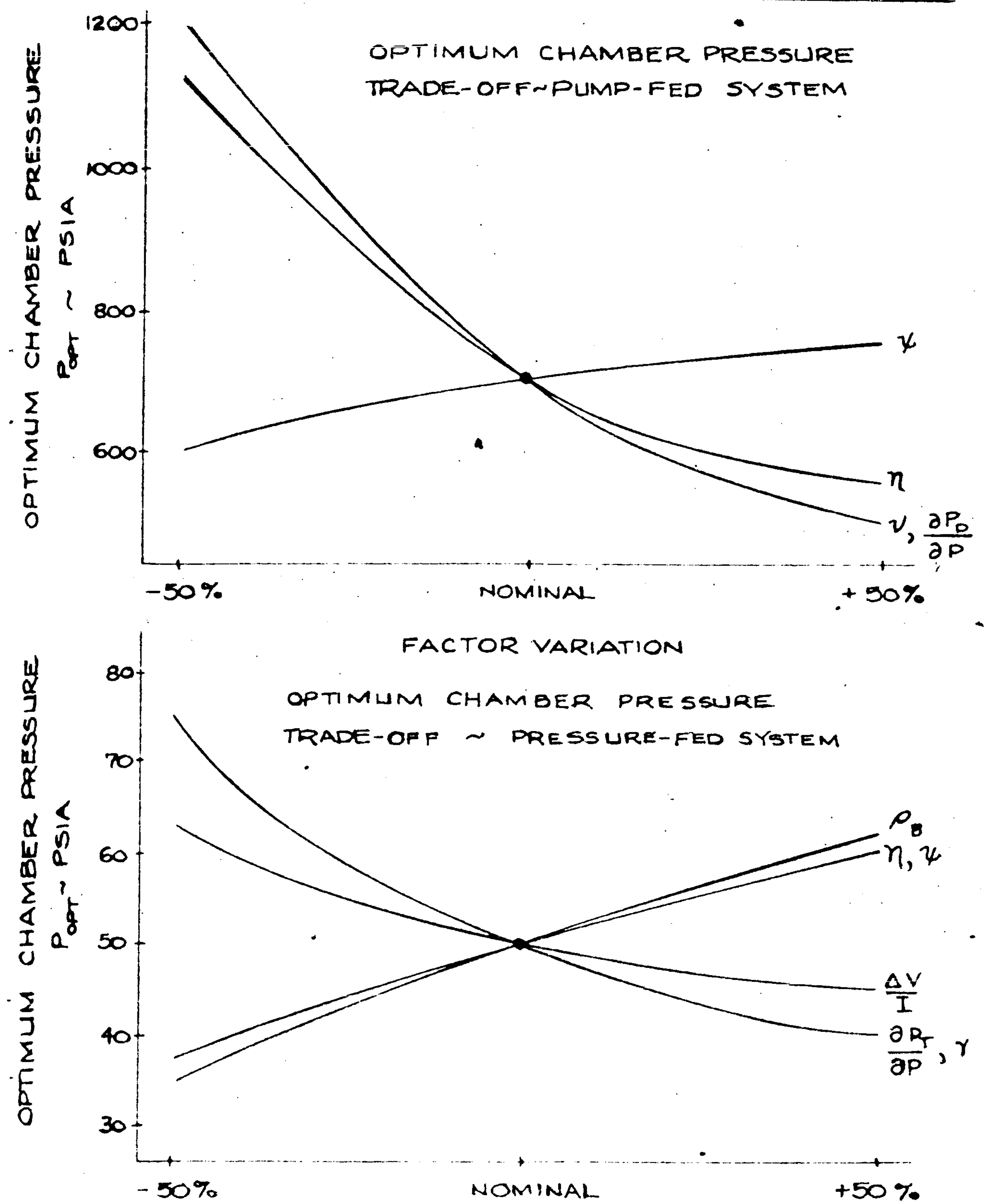

Figure 1-4. Factor Variation

[redacted]

TRANSIENT ENGINE PERFORMANCE

During engine start-up and cut-off, thrust is a function of time. This transient thrust build-up or decay contributes a certain amount of impulse to the vehicle. Due to variations in engine components this impulse contribution will vary from run to run in a given engine. These variations are illustrated in Fig. 1-5. This variation in impulse can significantly affect the trajectory to be traveled by the space vehicle and must be reduced to a negligible effect through engine system design and/or corrected in a subsequent propulsion phase.

For most propulsive maneuvers the variations in engine start can be taken into account by the guidance system during normal engine operation and the necessary correction made. Impulse deviations at cut-off were estimated for several propulsion systems as a function of engine thrust. Results for an LO_2/LH_2 engine are presented in Fig. 1-6. The deviation can be decreased by either lowering thrust or reducing the main valve closing time. There are, naturally, limits on both of these methods.

Some estimates were made of the effect of this cut-off impulse on the velocity of the space vehicle. These effects were considered in the lunar landing vehicles and the velocity variation was less than $\pm$ 1 fps.

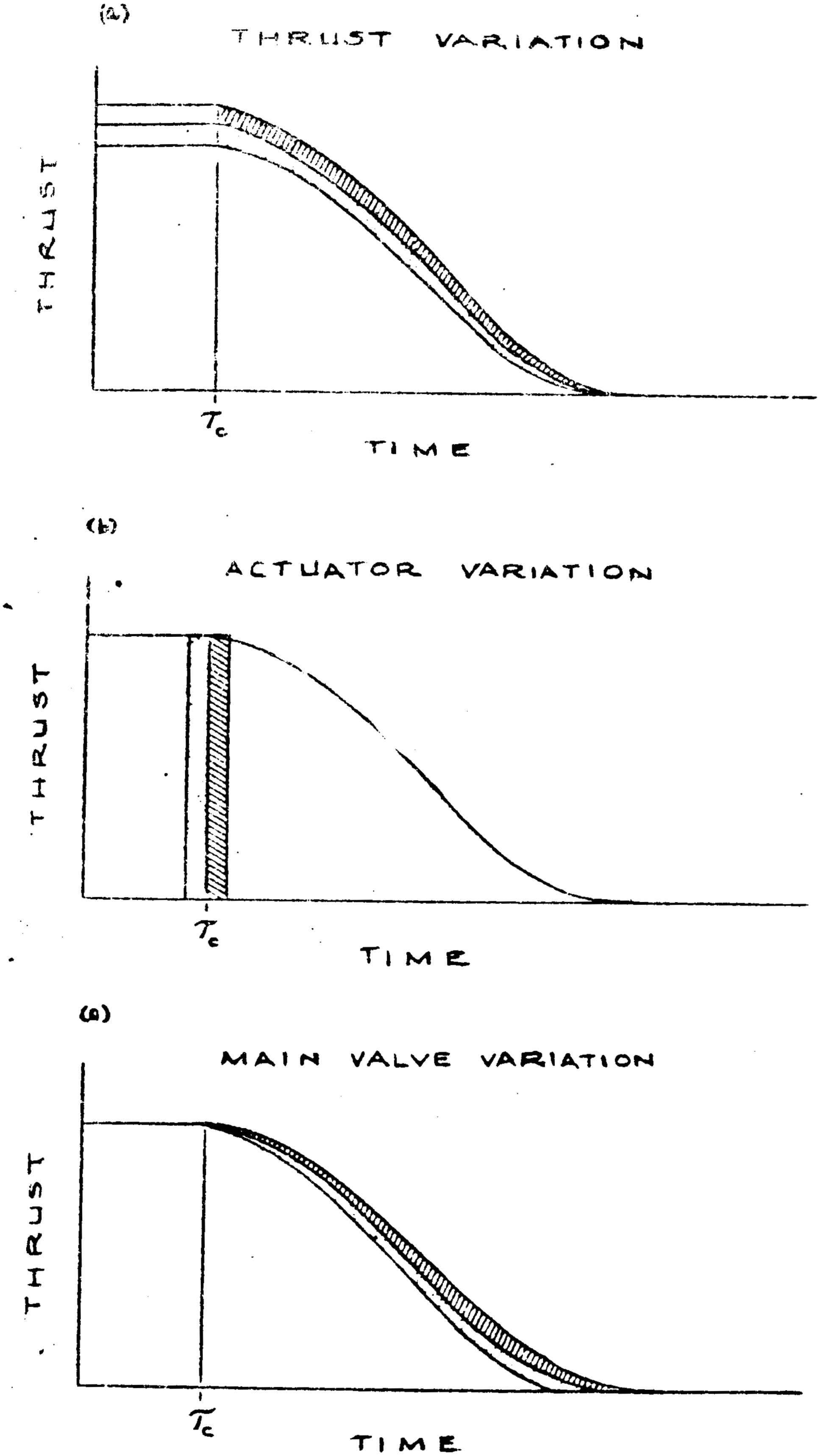

Figure 1-5. Variations in Cutoff Impulse

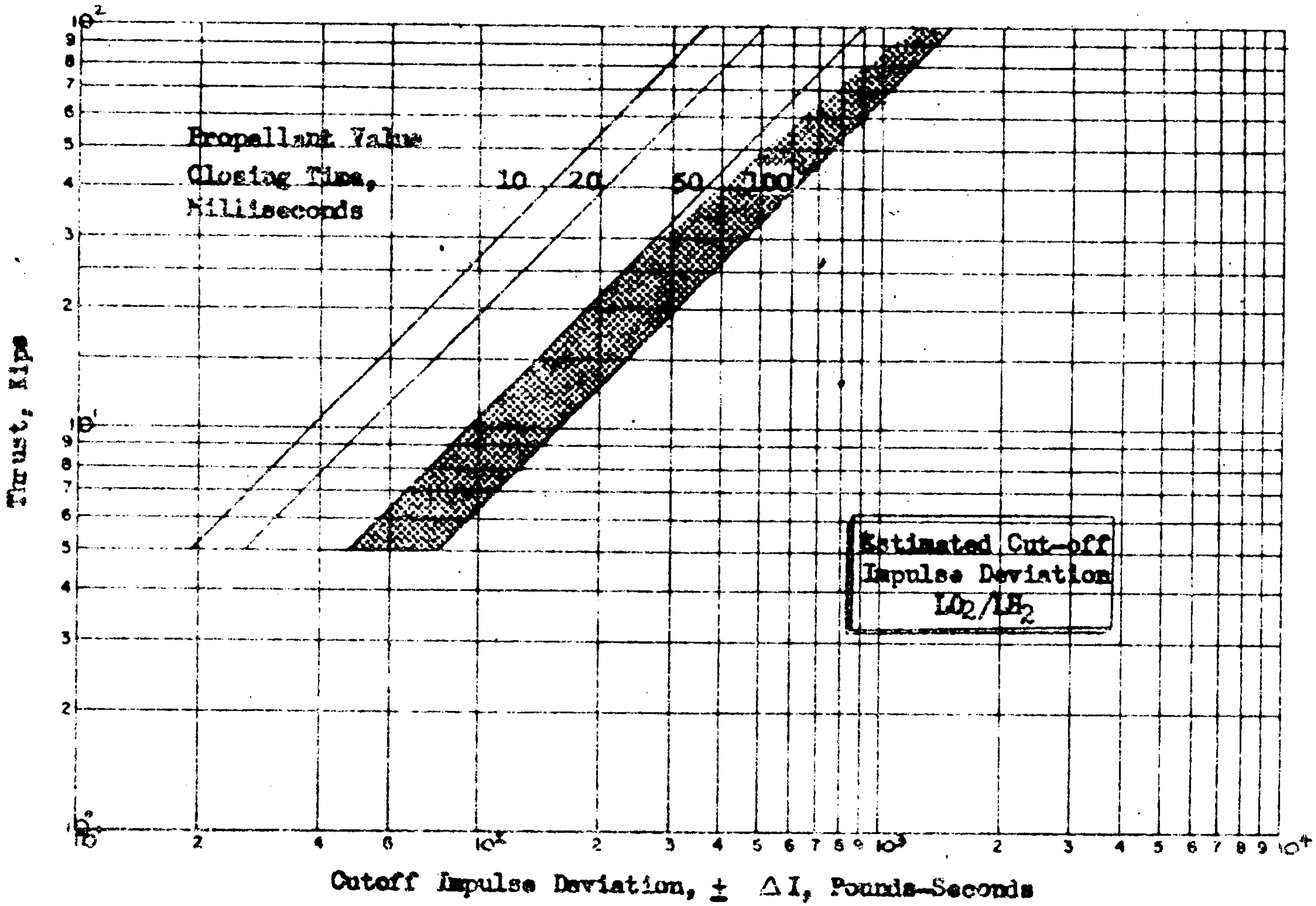

Figure 1-6. Estimated Cutoff Impulse Deviation LO_2/LH_2

[redacted]

ENGINE THROTTLING

In several propulsive maneuvers the requirement for engine throttling may arise. This throttling may be accomplished in a single step or be continuous. The performance losses accompanying throttled operation were studied. In general it was found that the losses in thrust chamber performance (specific impulse) were small for space applications. For example, a case of 5:1 throttling resulted in less than 2% decrease in thrust chamber specific impulse at the throttled condition. In a pump-fed system the decrease in engine specific impulse may be greater due to inefficiencies in turbine operation.

SECONDARY PROPULSION SYSTEM CONSIDERATIONS

In addition to the topics discussed above there are some secondary aspects which were considered. Trapped propellant, propellant utilization system, thrust vector control requirements, and vehicle acceleration loads all affect the propulsion system design.

A study was made of the trapped propellant in a LO_2/LH_2, pump-fed vehicle. Trapped propellant is that propellant that is left unburned due to premature exhaustion of the other propellant. Taking into account off mixture ratio tanking and deviation from expected time-average engine mixture ratio operation, the trapped propellant was estimated to be 1.38% of the useable propellant based on a fuel bias propellant tanking.

[redacted]

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[redacted]

For propulsion systems which supply fairly large velocity increments this amount of trapped propellant will significantly decrease the payload capability. In these systems a mixture ratio control or propellant utilization system would be beneficial.

Analysis of thrust vector corrective torque requirements for space vehicles indicates that generally a gimbal angle of 1 to 2 degrees (together with an auxiliary roll control system if a single engine is used) should be employed. Although for some space powered-flight maneuvers a small separate attitude control system would be adequate (thereby allowing a non-gimballed engine) consideration of engine thrust vector and vehicle c.g. misalignments dictates main engine gimbaling. For a specific vehicle these requirements should be analyzed in more detail so that vehicle dynamics can be considered.

Study of the acceleration loads to which a space vehicle would be subjected indicates that the inherent low initial thrust-to-weight systems required would result in low flight loads during space stage operations: approximately 4 g's axial and 0.5 g's lateral. More severe requirements are dictated by boost phase and ground handling considerations. Nominal values for these effects are:

Direction	Load	Operation Phase
Axial	8 g's	Boost
Lateral	4 g's	Handling

[redacted]

SPACE PROPULSION SYSTEM SPECIFICATION CATALOG

A catalog of specifications which will characterize the space propulsion system was developed from information acquired during this study and from previous Rocketdyne experience: This catalog is divided into two sections. The first, Table 1-21, presents general propulsion system information while the second, Table 1-12, is concerned with the system components.

[redacted]

TABLE 1-21

SPACE PROPULSION SYSTEM SPECIFICATIONS

GENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

A. Total Impulse Required (or Ideal Velocity Increment)

(1) Maximum; Mission

(2) Minimum; Mission

B. Maximum Impulse (Velocity) Increment; Mission

C. Minimum Impulse (Velocity) Increment; Mission

D. Number of Increments

II. Thrust

A. Magnitude

(1) Steady-State Design Thrust Magnitude

(a) Thrust-to-Earth Weight Ratio

(b) Absolute Value

(2) Tolerance

(a) Engine-to-Engine

(b) Run-to-Run

(3) Throttling

(a) Step

(b) Continuous

[redacted]

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TABLE 1-21

(Continued)

(4) Accuracy of Thrust Programming

(5) Number of Restarts

(6) Type of Thrust Control

B. Transients

(1) Start Sequence (ignition and response time)

(2) Start-up Impulse

(a) Nominal

(b) Tolerance

(3) Cut-off Impulse

(a) Nominal

(b) Tolerance

(4) Throttling Transition and Response Time

(a) Step

(b) Continuous

C. Thrust Vector Control

(1) Vector Control Requirement

(2) Method of Control

(3) Engine Thrust Vector Misalignment

(a) angular

(b) lateral

TABLE 1-21

(Continued)

III. Propellants

A. Composition

B. Mixture Ratio

(1) Nominal

(2) Tolerance

(3) Mixture Ratio Range

C. Specific Impulse

(1) Reference engine parameters

(a) Mixture ratio

(b) Chamber pressure

(c) Expansion ratio

(2) Nominal specific impulse at reference conditions

(3) Minimum at reference conditions

(a) Run to Run

(b) Engine to Engine

D. Compatibility with manned missions

E. Contamination effects on alien environment

F. Temperature Effects

(1) Density

(2) Vapor Pressure

(3) Heat of Vaporization

(4) Heat of Fusion

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TABLE 1-21

(Continued)

IV. Environmental Restrictions

A. Zero gravity propellant supply

(1) Liquid/vapor separation requirement

(2) Number of zero gravity engine starts

(3) Separation method

(4) Tank venting

(a) requirement

(b) method

B. Space Storage of Propellants

(1) Environment

(a) Thermal Radiation

(b) Internal Heat Source

(c) Ionizing Radiation

(d) Meteoroids

(2) Storage Time

(3) Propellant Temperature Limits

(4) Storage Methods

(a) exposed surface characteristics

(b) external insulation

(c) internal design

(d) propellant boil-off

(e) meteoroid shield

(f) deleterious effects of environment on storage methods

(g) altitude and geometry limits

TABLE I-21

(Continued)

C. Component Design Restrictions

(1) Meteoroids - puncture

(2) Temperature

(3) Ionizing Radiation-materials

(4) Vacuum-materials, start

D. Launch Environment

(1) Thermal

(2) Handling

E. Target or Payload Contamination

(1) (a) Bacteriological (Living)

(b) Chemical (Non-living)

(2) Decontamination Methods

(a) Cleaning

(b) Sterilization

(3) Con[illegible]

F. System Purging Requirements

(1) Number of Purges

(2) Type of Gas

(3) [illegible]

V. System Reliability [illegible]

A. Component

B. Engine

C. Vehicle

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TABLE 1-21

(Continued)

VI. Off Design Operation

A. Exchange Factors for Perturbation from Nominal

(1) Engine Operating Parameters

(a) Mixture Ratio

(b) Chamber Pressure

(c) Expansion Ratio

(d) Thrust

(2) Hardware Weight Equivalent of Specific Impulse

B. Alternate Mission Performance

TABLE 1-22

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants

b. Nominal Mixture Ratio

c. Propellant Temperature Limits

2. Useable Propellant

a. Maximum

b. Minimum

3. Reserve Propellant Weights

a. Flight Performance

b. Residual

c. Trapped

d. Fuel Bias

e. Boil-off Reserve

B. Tank Loads

1. Handling

2. Launch

3. In Atmosphere Flight

4. Space Flight

TABLE 1-21

(Continued)

C. Tank Pressure and NPSH

1. Nominal

2. Tolerance

D. Thermal Control

1. Thermal Loads

a. Ground Loads

b. Aerodynamic

c. Internal

d. Space Loads

2. Temperature Limits

E. Zero Gravity Requirements

1. Gas/Liquid Separation Requirements

2. Tank Venting Requirement

F. Staging, Geometry, and Configuration Requirements

G. Secondary Auxiliary Systems

H. Propellant Utilization System Requirements

I. Tank and Structure Weight Limits

II. Pressurization System

A. Purposes of Pressurization

B. Gas Volume in Propellant Tank

1. Total

2. Increments

TABLE 1-21

(Continued)

C. Gas Pressure in Propellant Tank

1. Nominal

2. Tolerance

D. Propellant Properties

1. Thermodynamic

2. Compatibility

E. Environment

1. Storage

a. Time

b. Gas Volume During Storage

2. Thermal Environment

3. Zero Gravity

F. Weight

III. Engine System

A. Propellant Description

1. Propellants

2. Thermodynamic Properties

3. Mixture Ratio

a. Nominal

b. Tolerance

B. Thrust

1. Nominal

2. Tolerance

3. Transients

[redacted]

TABLE 1-21

(Continued)

C. Type of Feed System

D. Specific Impulse

1. Nominal

2. Minimum

E. Engine Inlet Conditions

1. Storage

a. Time

b. Gas Volume During Storage

2. Thermal Environment

3. Zero Gravity

F. Weight

III. Engine System

A. Propellant Description

1. Propellants

2. Thermodynamic Properties

3. Mixture Ratio

a. Nominal

b. Tolerance

B. Thrust

1. Nominal

2. Tolerance

3. Transients

C. Type of Feed System

[redacted]

[redacted]

TABLE 1-21

(Continued)

D. Specific Impulse

1. Nominal

2. Minimum

E. Engine Inlet Conditions

F. Envelope Requirements

G. Throttling Requirements

1. Step

2. Continuous

H. Engine System Weight

I. Environment

[redacted]

[redacted]

LUNAR LANDING AND RETURN MISSION

In the study of the lunar landing and return mission, a large variety of propulsive maneuver combinations were considered. The various schemes considered are described in Fig. 1-7 through 1-10. From this maneuver spectrum, two basic methods of accomplishing the lunar mission evolved: (1) direct lunar landing (Fig. 1-8) and (2) orbital landing using an intermediate lunar orbit (Fig. 1-9). These two methods are illustrated in the figures by the shaded regions.

As in all of the space missions, the space vehicle was assumed to be initiated in a 300 n mi Earth orbit. The space vehicle departs from the orbit with a propulsion phase that has the thrust vector aligned with the velocity vector. This phase terminates when the vehicle has attained the energy required to make the Earth/Moon transfer in the desired time interval. Mid-course trajectory corrections are considered to be applied in two or more increments to reduce the landing error (CEP).

The direct 2 lunar landing method uses a maneuver which places the vehicle on the lunar surface directly from the transfer trajectory. This is accomplished by means of a fixed thrust level-interrupted burning maneuver in which the velocity and thrust vectors are essentially vertical during the firing. No substantial hovering or lateral translation provisions were included.

[redacted]

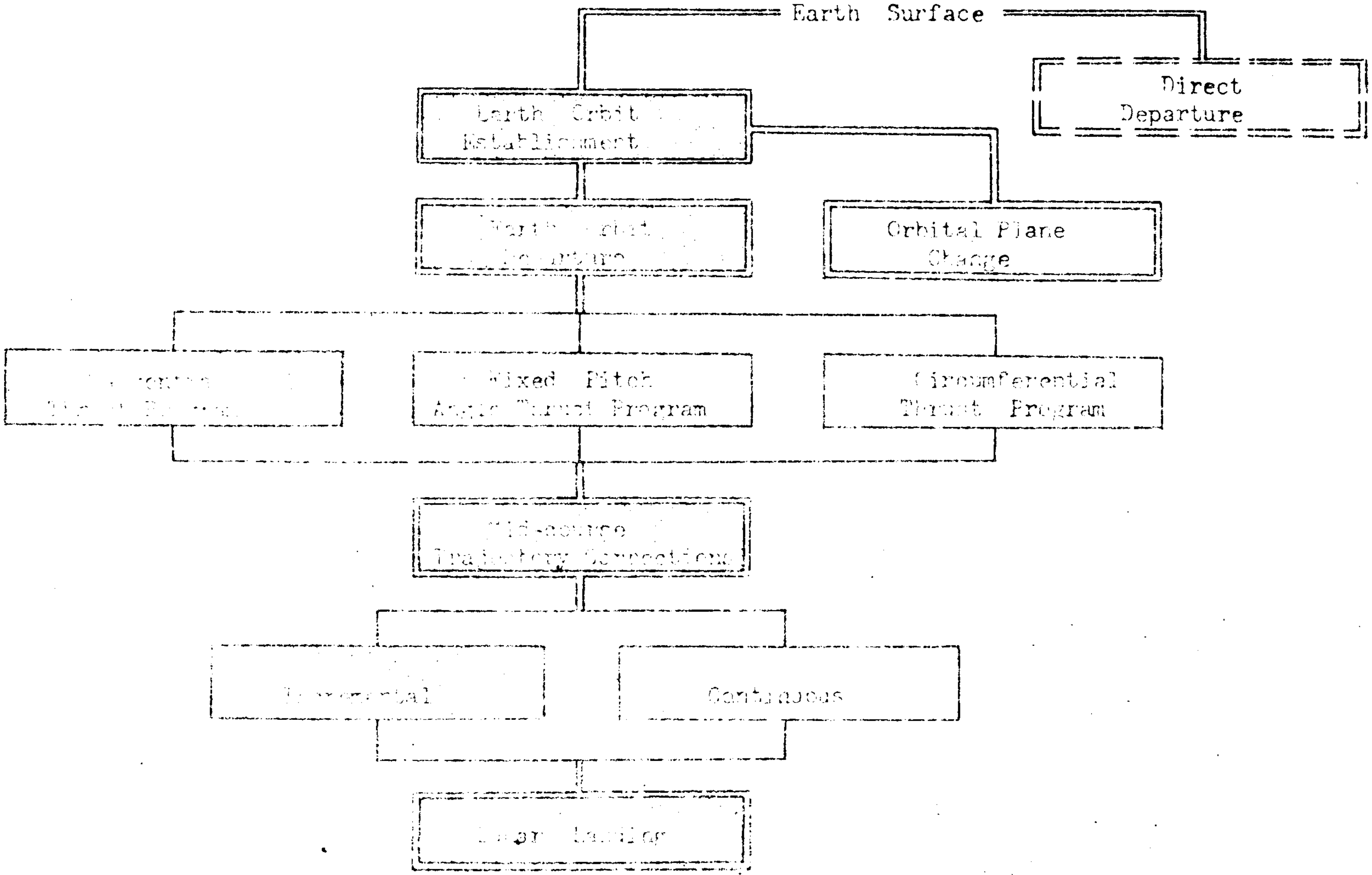


Figure 1-7. Maneuver Combinations for Earth-Moon Transfer

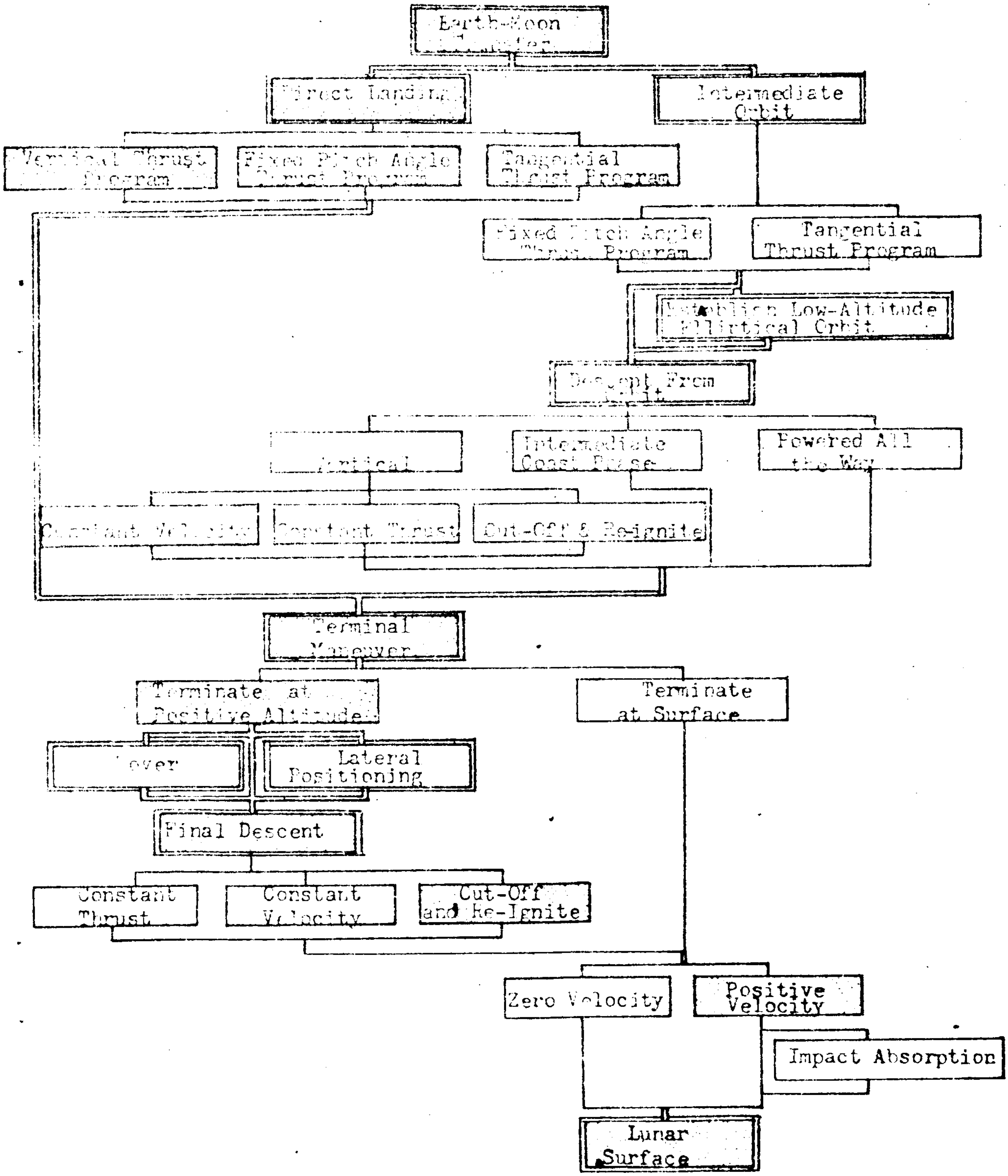


Figure 1-8. Maneuver Combinations for Lunar Landing

[redacted]

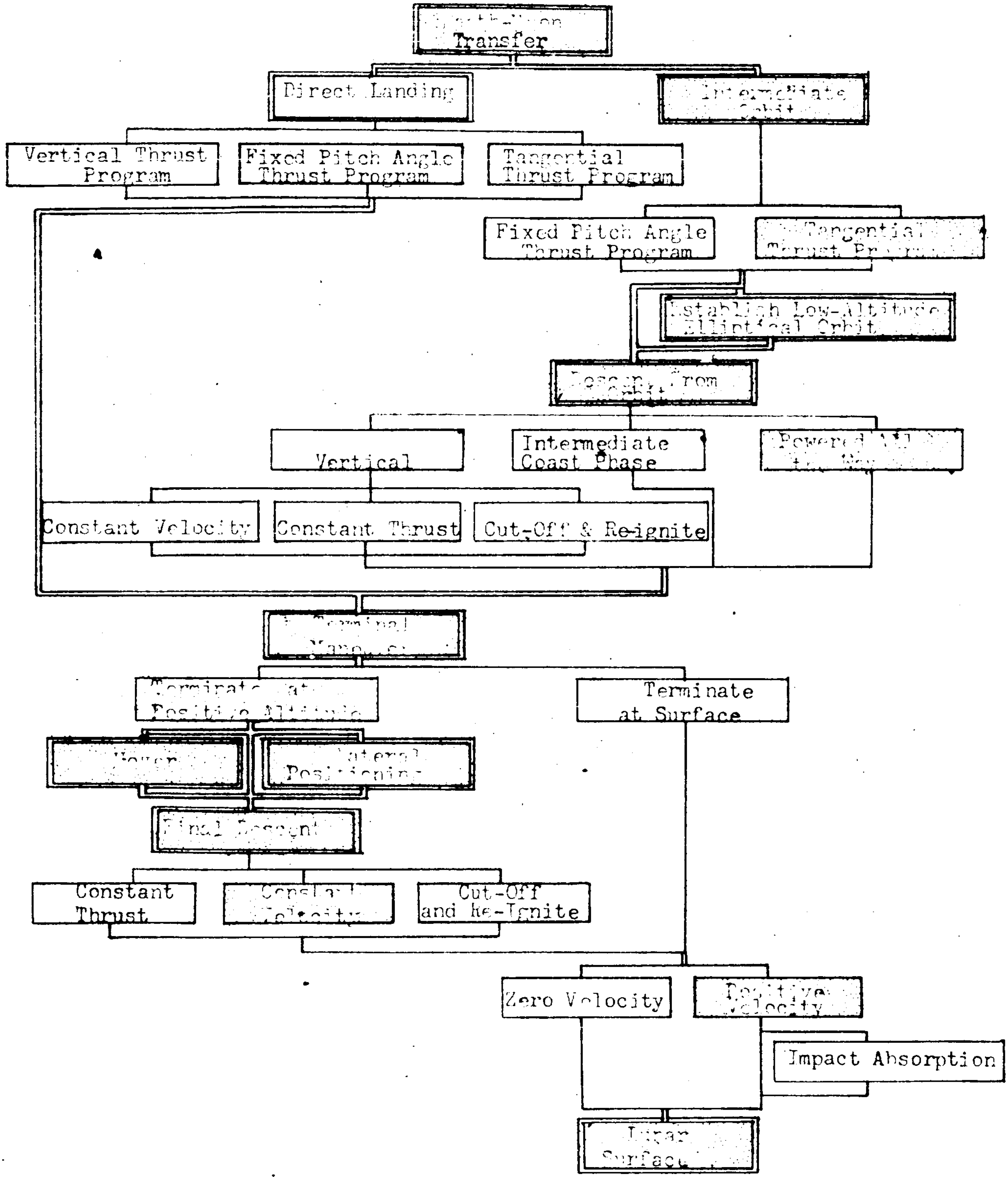


Figure 1-9. Maneuver Combinations for Lunar Landing

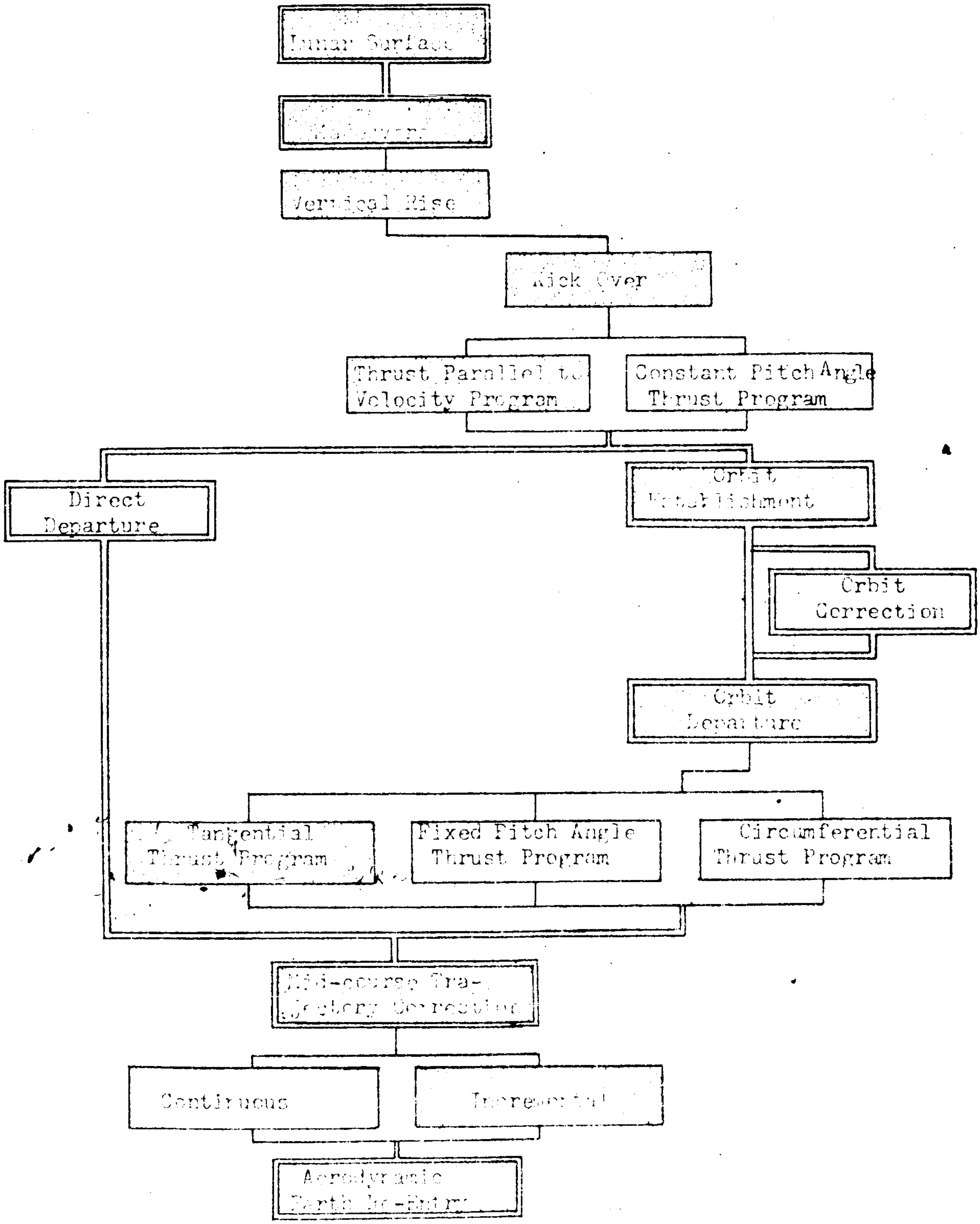


Figure 1-[illegible]. Maneuver Combinations for Moon-Earth Transfer

The lunar landing from orbit was also studied and is considered to be the most generally desirable type landing. A 50 n. mi. orbit is first established (the plane of which is determined by the velocity vector at the beginning of the transfer phase) using a thrust anti-parallel to velocity maneuver. This circular orbit is converted to a 50 n. mi./30,000 ft elliptical orbit with periopsis slightly before the desired landing spot. At the periopsis thrust is again applied anti-parallel to velocity to bring the vehicle to a low altitude with a small residual descent velocity. Hovering and translational capabilities are provided for this mission.

The takeoff maneuver was determined by the type of landing. Vertical takeoff to moon/earth transfer trajectory was used in conjunction with the vertical landing maneuver, and a takeoff to a 50 n. mi. orbit prior to the transfer was considered for the orbital landing case.

Midcourse corrections were provided for as in the earth/moon transfer trajectory. Earth re-entry and landing maneuvers were assumed to be accomplished aerodynamically.

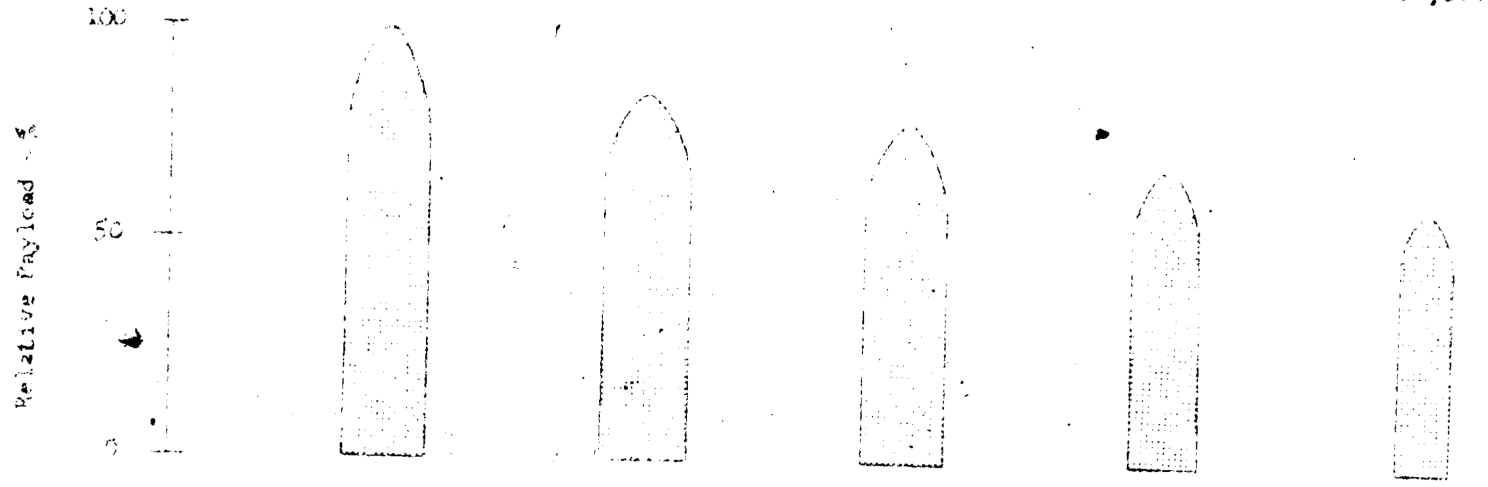
The vertical descent trajectory is most suited to systems having simple guidance systems and fixed thrust engines. The probability of safe return appears lower than that of the orbital trajectory and the landing point is restricted. Capability of one restart will be required. The orbital landing technique assures that the vehicle will not crash if the engines fail to ignite. The maneuver makes use of (assumed) previous lunar orbital experience and permits landing at any point on the lunar surface.

For the two maneuver combination methods, propulsion systems were studied to evaluate engine thrust levels, vehicle staging, and relative payload capability. These studies were based upon a space vehicle weighing 354,000 lbs, initially placed in a 300 n mi orbit by a NOVA H-6 booster vehicle. The effects of using four different propulsion systems were studied. These systems, liquid oxygen/liquid hydrogen (LO_2/LH_2) and mixed oxides of nitrogen/monomethylhydrazine (MON/MMH), represent a broad range of propulsion system characteristics and should indicate the effects of propellant properties on the space vehicle.

The propulsion systems were used in a variety of maneuver/vehicle combinations and were evaluated in terms of performance, complexity, etc. For example, the earth/moon transfer maneuver is considered using the previously mentioned systems plus a typical solid propellant system. A comparison of these vehicles, shown in Figure 1-11, clearly indicates the advantages of the LO_2/LH_2, pump-fed system.

Initial Gross Weight in 300 n. mile Earth Orbit = 354,000 lbs.
(Based on NOVA H-6 Capability)

Payload lbs	153,900	131,500	121,100	109,000	90,500



Propellants	LO_2/LH_2	LO_2/LH_2	MON/MMH	MON/MMH	SOLID
Feed System	Pump	Pressure	Pump	Pressure	--
Thrust ~ lbs	125,000	83,500	137,000	97,000	250,000
λp (Calc.)	0.927	0.876	0.950	0.912	0.900

Figure 1-11. Earth Moon Transfer Propulsion 2.6 Day Coast Time

[redacted]

From consideration of a large number of combinations the vehicles recommended for use in the two lunar missions were selected. These are described in Tables 1-23 and 1-24. Pump-fed systems were selected over the pressure-fed systems as they provide a significant payload advantage due to lighter engines and tanks. Propellant storage studies indicated that the LOX/H_2 combination could be easily maintained for the lunar missions contemplated. Thrust levels (Figure) are near optimum for the two-stage combination selected. However, a wide variation in thrust is possible without severe payload loss, e.g. the J-2 engine (200K) may be used as the first stage propulsion system in either vehicle without affecting payload appreciably.

The two-stage vehicles were selected since it was felt that their simplicity was of more benefit than the slight payload increases achieved with a greater number of stages. Other considerations, such as the desire to provide abort capability at all times during the mission, could modify the staging selection.

Staging on the lunar surface is preferable for the vertical descent trajectory because it permits use of a previously fired propulsion system for the critical landing phase, and has the added advantage of protecting the takeoff propulsion system from impact damage

[redacted]

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[redacted]

TABLE [illegible]-23

LUNAR LANDING AND RETURN VEHICLE

USING AN INTERMEDIATE LUNAR ORBIT

Payload Weight on Moon/Earth Transfer, lb	29,500

Stage Two

Propellants	oxygen/hydrogen
Feed System	Pump
Throttling	[illegible] [illegible] percent Continuous
Restarts	[illegible]
Gross Weight, lb	[illegible]26,000
Thrust, lb	[illegible]
Number of Engines	[illegible]
1 [illegible]	
1 Throttleable	

Stage One

Propellants	[illegible]
Feed System	[illegible]
Restarts	[illegible]
Gross Weight, lb	[illegible]
Thrust, lb	[illegible]
Number of Engines	[illegible]

* Maneuver performed by [illegible]

[redacted]

TABLE I-24

RECOMMENDED SYSTEM FOR DIRECT LUNAR LANDING AND RETURN MISSION

Payload Available For Earth Reentry (lb)	26,300

Stage Two

Propellants	LO_2/LH_2
Feed System	Pump
Restarts	None
Gross Weight (lb)	37,500
Thrust (lb)	56,000

Stage One

Propellants	LO_2/LH_2
Feed System	Pump
Restarts	1
Gross Weight (lb)	354,000
Thrust (lb)	248,000

upon touchdown. For the orbital landing, it is desirable to stage prior to the descent from orbit maneuver in order that the thrust required for that maneuver does not unfavorably influence the thrust level selection of previous maneuvers. Preliminary review tends to indicate a redundant multi-engine propulsion system should be used for increased landing reliability.

A broadband (5°), three axis, attitude control, propulsion system which functions during the entire transfer can be included at a weight of less than 100 lb. The midcourse correction and orbital conversion maneuvers can be performed by the main propulsion system using the attitude control engines for propellant settling.

A description of the propulsion systems used in the lunar land and return mission (intermediate orbit method) is presented in Tables 1-25 and 1-26. These descriptions were developed based on the propulsion system specification catalog.

[redacted]

TABLE 1-25

SPACE PROPULSION SYSTEM SPECIFICATIONS

Lunar Landing and Return Vehicle (Orbital): Stage 1

GENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

- A. Total Ideal Velocity Increment Required = 13,700 fps
- B. Maximum Velocity Increment
  - 1. Increment = 10,150 fps
  - 2. Mission: Earth/Moon Transfer
- C. Minimum Velocity Increment
  - 1. Increment = 150 fps
  - 2. Mission: Mid-course Correction
- D. Number of Increments = 4

II. Thrust

- A. Magnitude
  - 1. Steady-state Design Thrust Magnitude
    - a. Initial Thrust-to-Earth Weight Ratio = 0.35
    - b. Absolute Value = 125,000 lb
  - 2. Tolerance
    - a. Engine-to Engine: ± 3%
    - b. Run-to-Run: ± 1%
  - 3. Throttling
    - a. Step: None
    - b. Continuous: None

[redacted]

TABLE 1-25

(Continued)

4. Number of Restarts: 3

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

1. Nominal: 5.0 (O/F)

2. Tolerance: ± 0.5 percent

C. Specific Impulse

1. Reference Engine Parameters

a. Mixture Ratio: 5.0 (O/F)

b. Chamber Pressure: 500 psia

c. Expansion Ratio: 30

2. Nominal Engine Specific Impulse at Reference Conditions: 428 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start.

2. Number of Zero Gravity Engine Starts: 4

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth/Moon Vicinity

2. Storage Time: 4-5 days

[redacted]

TABLE 1-25

(Continued)

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

C. Component Design Restrictions: Protect from, or Design for, Earth/Moon Vicinity Space Environment.

F. System Purging Requirements:

1. Number of Purges: 3

[redacted]

TABLE 1-25

(Continued)

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5.0 (O/F)

2. Useable Propellant Weight: 223,150 lb

3. Reserve Propellant Weight

a. Flight Performance: 2,230 lb

c. Trapped: 3, 060 lb

d. Fuel Bias: 1,225 lb

e. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 3 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for [illegible] engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purposes of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

TABLE 1-25

(Continued)

B. Gas Volume in Propellant Tank

1. Increments: 4

E. Environment

1. Storage

a. Time: 4 to 5 days

2. Thermal: Earth/Moon Vicinity

III. Engine System

A. Propellant Description

1. Propellants: Liquid Oxygen/Liquid Hydrogen

3. Mixture Ratio

a. Nominal: 5.0 (O/F)

b. Tolerance: ± 0.5 percent

B. Thrust

1. Nominal: 125,000 lb

2. Tolerance

a. Run-to-Run: ± 1 percent

b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Turbopump

D. Specific Impulse

1. Nominal: 428 sec

G. Throttling Requirement

1. Step: None

2. Continuous: None

I. Environment: Earth/Moon Vicinity Space

[redacted]

TABLE 1-26

SPACE PROPULSION SYSTEM SPECIFICATIONS

Lunar Landing and Return Vehicle (Orbital): Stage 2

GENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

A. Total Ideal Velocity Increment Required = 15,770 fps

B. Maximum Velocity Increment

1. Increment = 6400 fps

2. Mission: Landing from Orbit

C. Minimum Velocity Increment

1. Increment = 60 fps

2. Mission: Elliptical Orbit Establishment

D. Number of Increments = 6

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.68

b. Absolute Value = 77,500 lb

2. Tolerance

a. Engine-to-Engine: ± 3 percent

b. Run-to-Run: ± 1 percent

3. Throttling

a. Step: 6:1

b. Continuous: 6 percent

[redacted]

TABLE 1-26
(Continued)

4. Number of Restarts: 5

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

1. Nominal: 5.0 (O/F)

2. Tolerance: ± 0.5 percent

C. Specific Impulse

1. Reference Engine Parameters

a. Mixture Ratio: 5.0 (O/F)

b. Chamber Pressure: 500 psia

c. Expansion Ratio: 30

2. Nominal Specific Impulse at Reference Conditions: 428 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start

2. Number of zero gravity engine starts: 5

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth/Moon Vicinity

2. Storage Time: 2 Weeks

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

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TABLE 1-26

(Continued)

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

C. Component Design Restrictions: Protect from, or Design for, Earth/Moon Vicinity Space Environment.

F. System Purging Requirements:

1. Number of Purges: 5

TABLE 1-26

(Continued)

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5.0 (O/F)

2. Useable Propellant Weight: 76,150 lb

3. Reserve Propellant Weight

a. Flight Performance: 762 lb

c. Trapped: 1,040 lb

d. Fuel Bias: 418 lb

e. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 3 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for 5 engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purposes of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

TABLE 1-26

(Continued)

B. Gas Volume in Propellant Tank

   1. Increments: 6

E. Environment

   1. Storage

      a. Time: 2 Weeks

   2. Thermal: Earth/Moon Vicinity

III. Engine System

A. Propellant Description

   1. Propellants: Liquid Oxygen/Liquid Hydrogen

   3. Mixture Ratio

      a. Nominal: 5.0 (O/F)

      b. Tolerance: ± 0.5 percent

B. Thrust

   1. Nominal: 77,500 lb

   2. Tolerance

      a. Run-to-Run: ± 1 percent

      b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Turbopump

D. Specific Impulse

   1. Nominal: 428 sec

G. Throttling Requirement

   1. Step: 6:1

   2. Continuous: 6 percent

I. Environment: Earth/Moon Vicinity Space

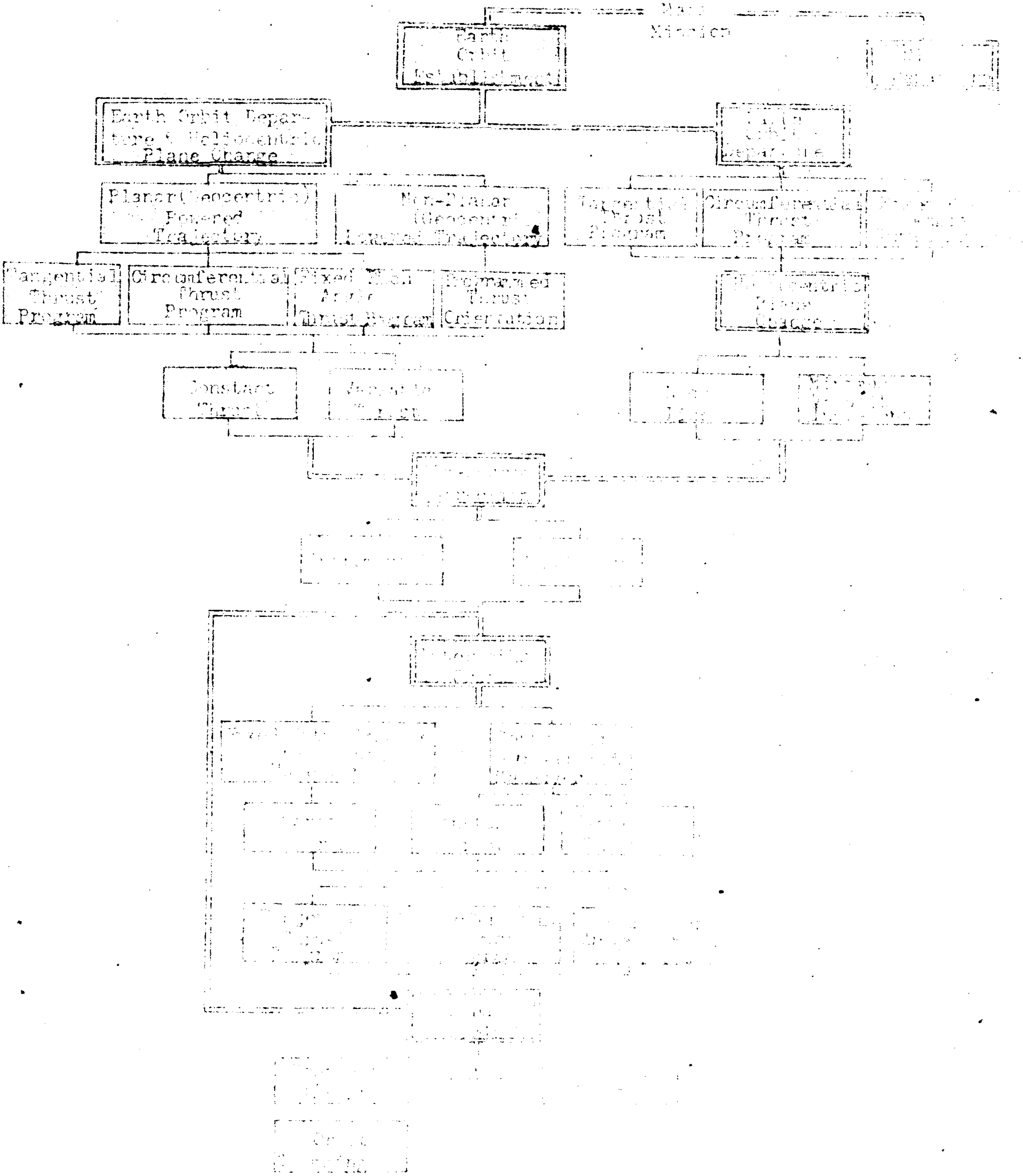
MARS ORBIT ESTABLISHMENT MISSION

The recommendations and conclusions of the Mars Orbit Mission studies of Phase 2 can be divided into two categories: Those conclusions relative to the propulsion/vehicle system, and those pertaining to maneuvers. The separation does not imply independence of the parameters within the broad categories.

A variety of propulsive maneuver combinations were considered for this mission. These are indicated in Figure 1-12. The maneuver combinations selected after analysis and review, for accomplishing this mission are indicated by the shaded areas in this figure. Analysis of Earth-Mars interplanetary trajectories, based on simulated elliptical planetary orbits, indicate a minimum energy launch period occurs about once every two years, and results in Earth-Mars transfer times of approximately 170-240 days dependent upon the year of launch.

During a minimum energy period the space vehicle will depart from an earth orbit inclined to the equator to permit a planar propulsion phase. The thrust vector is aligned with the velocity vector during the propulsion phase. The propulsion phase terminated after the vehicle has attained the energy requirements of the particular launch date.

[redacted]



[redacted]

The mid-course corrections are to be applied in two increments. The first is applied after a 20 day delay from launch and the second shortly prior to planetary intercept. The second or terminal correction for establishing the entry corridor (entry corridor correction) provides the desired asymptotic approach distance at Mars. Both corrections modify the trajectory to maintain a constant transfer time.

A Mars capture maneuver that employs the pre-established asymptotic approach distance is recommended. The retro-thrust propulsion maneuver initiates at an altitude determined by the hyperbolic approach velocity of the vehicle with respect to the planet. An intermediate orbit is established to ensure capture by the planetary gravitational field. The intermediate orbit is corrected by Hohmann type maneuvers to the final recommended 300 n. mi. circular orbit.

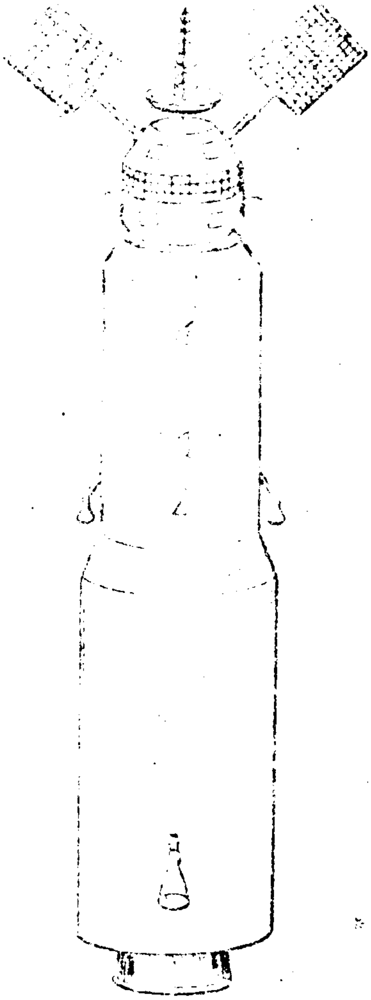
Propulsion system studies were conducted for this mission using a 354,000 lb space vehicle placed in a 300 n. mi. earth orbit by a NOVA H-6 booster vehicle. The natural velocity increment separations, and storage periods involved indicate that a separate stage for each of the two major propulsion phases (Earth and Mars) be employed. The space vehicle is designed to be capable of launch at anytime during a one-month interval of earth orbit departure dates which occur at approximately two-year intervals. The stages are filled with the propellant requirements of the launch date. The flexibility permits the basic vehicle to be applied to the optimum launch periods of a number of years. The space vehicle is described in Fig. 1-13.

The first stage for Earth-orbit departure uses liquid oxygen/liquid hydrogen as propellants. The propulsion system operates at a constant thrust level and is a pump-fed system. The thrust level for the nominal vehicle has been selected to be 150,000 lb thrust. Based on the engine grouping in Phase I of this study, the propulsion system of this stage could be two 75,000-lb-thrust engines. However, if preferred, the 200,000 lb thrust (J-2) O_2/H_2 engine presently under development could be used with very little performance change. This performance change is demonstrated in Fig. 1-15, which shows the effect of the first stage thrust magnitude upon the stage payload-to-gross weight ratio. The figure also shows the performance change for a selected thrust magnitude when the initial gross weight of the space vehicle is changed. This illustrates the application of the propulsion system to other vehicles.

For the nominal 354,000 lb vehicle, a second stage thrust level of 30,000 lb is recommended. Figure 1-17 presents the payload-to-gross weight ratio for the second stage as a function of the thrust magnitude and the initial gross weight. It shows the performance change of a

[redacted]

Gross Weight: 354,000 lbs.



Payload: [illegible] lbs.

Stage Two:*

Thrust: 30,000 lbs.

Propellants: LO_2/LH_2

Propellant Tanks:

	Design Capacity	Loading Variations
LO_2 :	[illegible] lbs.	[illegible] - [illegible],330 lbs.
LH_2 :	[illegible],550 lbs.	[illegible],550 - [illegible],550 lbs.

Turbopump Fed

Stage One:

Thrust: 150,000 lbs.

Propellants: LO_2/LH_2

Propellant Tanks:

	Design Capacity	Loading Variations
LO_2 :	39,250 lbs.	35,130 - 39,250 lbs.
LH_2 :	19[illegible],250 lbs.	175,[illegible] - 19[illegible],250 lbs.

Turbopump Fed

* Stage designed to establish intermediate orbit and to restart to change to final orbit.

Figure 1-13. Recommended Two-Stage Vehicle for Mars Orbit Mission

[redacted]

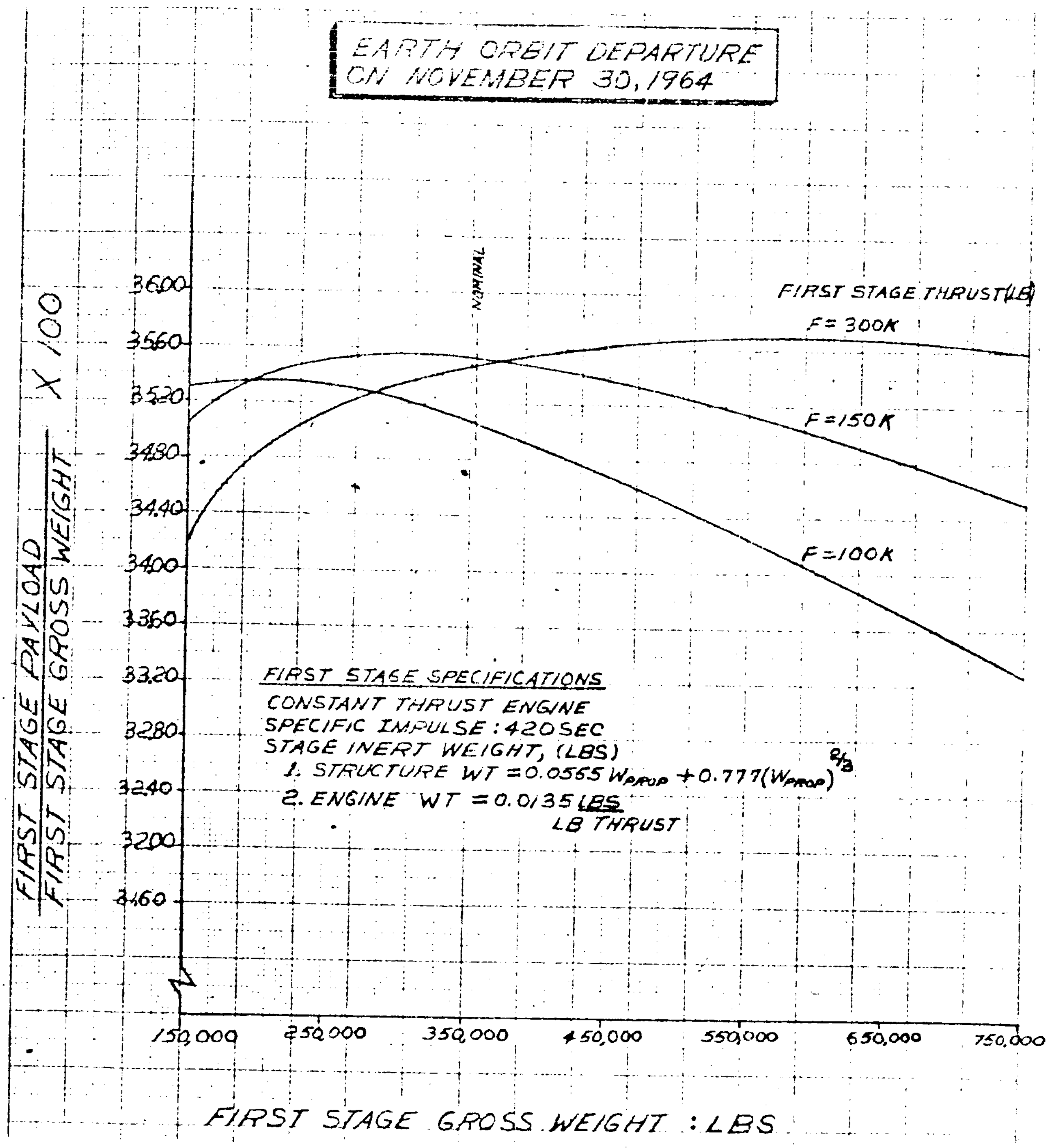


Figure 1-14. 200-Day Mars Transfer, First-Stage Specifications

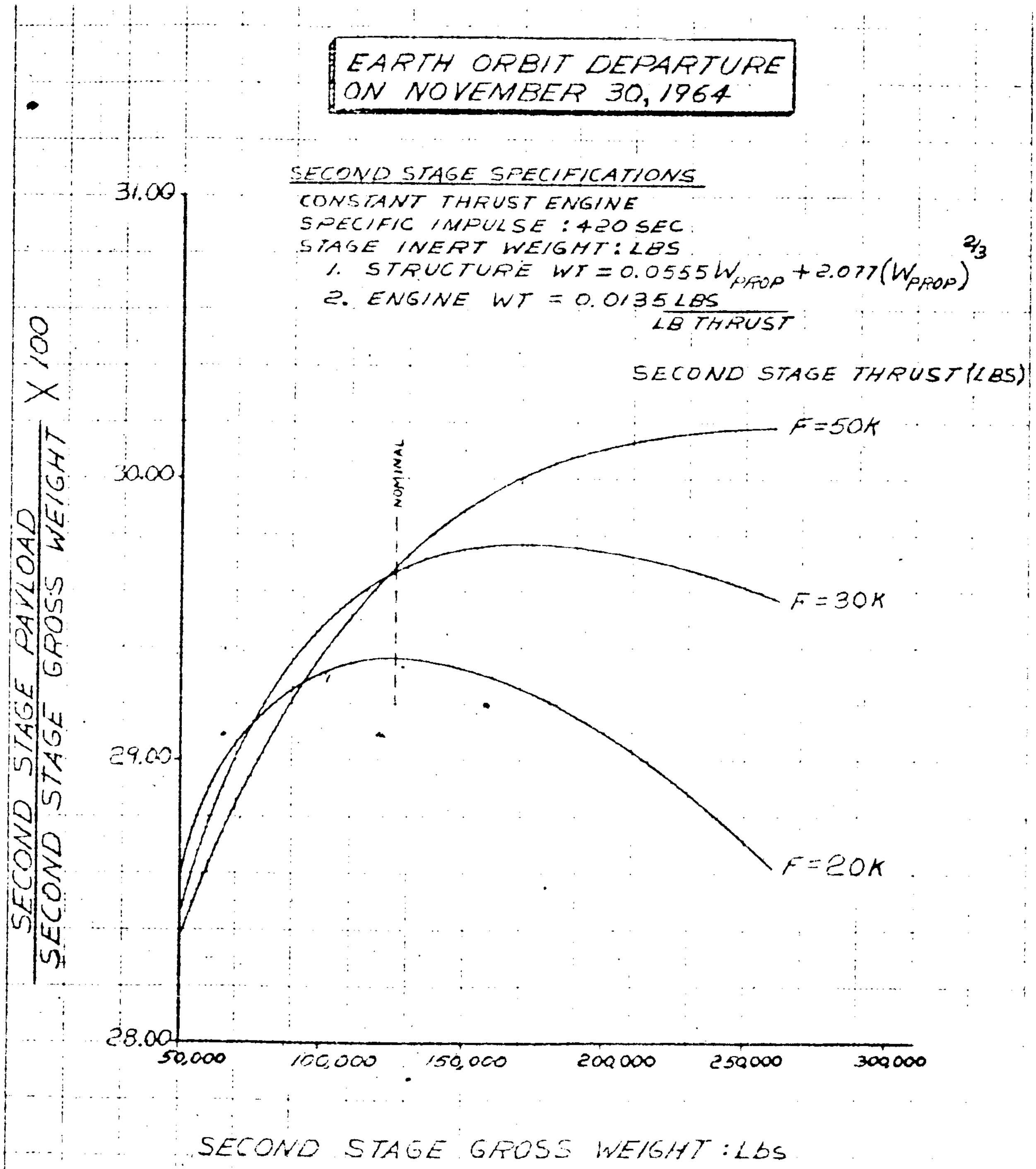


Figure 1-15. 200-Day Mars Transfer, Second-Stage Specifications

propulsion system operation with a change in the initial gross weight; and the performance change for a selected vehicle with a thrust magnitude change.

For this stage liquid oxygen/liquid hydrogen propellants are feasible. If the initial gross weight of the space vehicle decreases significantly, storable propellants appear more advantagous. As the initial gross weight of the space vehicle increases the cryogenic propellants definitely appear more favorable. The seoond stage propulsion system is recommended as a cluster of three-10,000 lbs thrust engines operating at a constant thrust level. One of the engines must be restartable to change the intermediate orbit into the final orbit. The propulsion systems will be pump-fed.

The mid-course corrections should be applied by an independent system with a capability of approximately 300 ft/sec total velocity increment. This system will be external to the sealed second stage engine system. The mid-course correction system may be an integral part of, or the totality of the system required for attitude control of the vehicle during the transfer phase. Since attitude control system analyses were beyond the scope of this study, no recommendations can be made as to the integration or separation of the mid-course propulsion system with the attitude control system.

[redacted]

A description of the propulsion systems used in this Mars mission is presented in Tables 1-27 and 1-28 based on the propulsion system specification catalog.

[redacted]

[redacted]

TABLE 1-27

SPACE PROPULSION SYSTEM SPECIFICATIONS

Mars Orbit Establishment Vehicle, First Stage

General Propulsion System Description

I. Energy Requirements

A. Total Impulse Required

1. Maximum = 9.8766×10^7 lb-sec.

2. Minimum = 8.850×10^7 lb-sec.

B. Maximum Impulse:

1. Increment = 9.8766×10^7 lb-sec.

2. Mission: Earth Orbit Departure

C. Minimum Impulse

1. Increment = 8.850×10^7 lb-sec.

2. Mission: Earth Orbit Departure

D. Number of Increments = 1

II. Thrust

A. Magnitude

1. Steady-State Design-Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.4237

b. Absolute Value = 150,000 lb.

[redacted]

TABLE 1-27

(Continued)

2. Tolerance

    a. Engine-to-engine: ±3.0 percent

    b. Run-to-run: ±1.0 percent

3. Throttling

    a. Step: None

    b. Continuous: None

4. Number of Restarts: 0

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

    1. Nominal: 5 (o/f)

    2. Tolerance: ±5 percent

C. Specific Impulse

    1. Reference Engine Parameters

        a. Mixture Ratio: 5 (o/f)

        b. Chamber Pressure: 500 psia

        c. Expansion Ratio: 30

    2. Nominal Specific Impulse at Reference Conditions: 428 sec.

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TABLE 1-27

(Continued)

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propella for engine start.

2. Number of Zero Gravity Engine Starts: 1

3. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth Vicinity

2. Storage Time: A few days

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limit of propellant tank and engi

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engi

C. Component Design Restrictions: Protect from or Design for Earth Vicinity Environment.

D. System Purging Requirements:

1. Number of Purges: None

[redacted]

TABLE 1-27
(Continued)

System Component Requirements

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5 (o/f)

2. Useable Propellant Weight:

a. Maximum: 235,500 lb.

b. Minimum: 211,780 lb.

3. Reserve Propellant Weight

a. Flight Performance: 2,355 lb. (maximum)

b. Trapped: 3,220 lb. (maximum)

c. Fuel Bias: 1,290 lb. (maximum)

d. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

2. Atmosphere Flight: 3 g Axial

3. Space Flight: 4 g Axial

C. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellant for engine start.

2. Tank Venting: None

[redacted]

TABLE 1-27

(Continued)

II. Pressurization System

A. Purposes of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

B. Gas Volume in Propellant Tank

1. Increments: 1

C. Environment

1. Storage

a. Time: a few days

2. Thermal: Earth vicinity

III. Engine System

A. Propellant Description

1. Propellants: Liquid Oxygen/Liquid Hydrogen

2. Mixture Ratio

a. Nominal: 5 (o/f)

b. Tolerance: ±0.5 percent

B. Thrust

1. Nominal: 150,000 lb.

2. Tolerance

a. Run-to-Run: ±1.0 percent

b. Engine-to-Engine: ±3.0 percent

[redacted]

TABLE 1-27

(Continued)

C. Type of Feed System: Turbopump

D. Specific Impulse

1. Nominal: 428 sec.

E. Throttling Requirement

1. Step: None

2. Continuous: None

F. Environment: Earth Vicinity

[redacted]

[redacted]

TABLE 1-28

SPACE PROPULSION SYSTEM SPECIFICATIONS

Mars Orbit Establishment Vehicle

General Propulsion System Description

I. Energy Requirements

A. Total Impulse Required

1. Maximum = 3.3993×10^7 lb-sec

2. Minimum = 2.0952×10^7 lb-sec

B. Maximum Impulse

1. Increment = 3.2907×10^7 lb-sec

2. Mission: Mars Intermediate Orbit Establishment

C. Minimum Velocity Increment

1. Increment = 0 lb-sec

2. Mission: Mars Intermediate Orbit Correction

D. Number of Increments = 3

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.2470 - 0.3093

b. Absolute Value = 30,000 lb

[redacted]

TABLE I-28

(Continued)

2. Tolerance

   a. Engine-to-engine: ± 3.0 percent

   b. Run-to-run: ± 1.0 percent

3. Throttling

   a. Step: 3:1

   b. Continuous: None

4. Number of Restarts: 2

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

   1. Nominal: 5 (O/F)

   2. Tolerance: ± 0.5 percent

C. Specific Impulse

   1. Reference Engine Parameters

      a. Mixture Ratio: 5 (O/F)

      b. Chamber Pressure 500 psia

      c. Expansion Ratio 30

   2. Nominal Specific Impulse at Reference Conditions: 428 sec

[redacted]

TABLE 1-28

(Continued)

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start. Possible venting requirement.

2. Number of zero gravity engine starts: [illegible]

4. Tank Venting: To relieve propellant heating problem.

B. Space Storage of Propellants

1. Environment: Earth-to-Mars vicinity

2. Storage Time: 250 days

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limit of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

[redacted]

TABIE 1-28

(Continued)

C. Component Design Restrictions: Protect from, or Design for Earth-to-Mars vicinity.

F. System Purging Requirements:

1. Number of Purges: 2

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EARTH ORBIT RENDEZVOUS MISSION

The maneuvers and staging employed for this orbital rendezvous mission are as follows: A conventional boost maneuver is accomplished by the first and second stage of the booster vehicle. The coast to apogee is followed by firing of the final stage to establish an orbit. Upon approaching the target the final stage is ignited to accomplish the plane change leaving a small residual closing velocity between the target and vehicle. Multiple on-off operation of the final stage is used to achieve rendezvous while the attitude control system is used to perform the actual docking maneuver. Aerodynamic re-entry, if required, is initiated by application of retro-thrust by final stage. This sequence appears attractive on the basis of reliability, guidance requirements, and payload considerations.

A restartable, fixed thrust level, pressure fed, storable propellant system having an F/W ratio in the order of 0.1 is recommended on the basis of reliability with consideration also given to payload capability and guidance requirements. The payload sensitivity to various propulsion systems is low due to the small velocity increment involved. The selected system is shown in Table 1-29 based on an H-2 booster and a 300 n mi orbital mission including a 5^{o} plane change. This system is further described in Table 1-30.

It is suggested that strong emphasis be placed on the operational aspects of the rendezvous mission since these may be a major factor in determining the mission characteristics.

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TABLE 1-29

RENDEZVOUS PROPULSION SYSTEM

Payload, lb	92,800
Propulsion System	
Feed System	Positive Expulsion
Propellants	MON/MMH
Propellant Weight, lb	28,400
Inert Weight, lb	3200
Thrust, lb	12,000
Restarts	3

[redacted]

TABLE 1-30

SPACE PROPULSION SYSTEM SPECIFICATIONS

Orbital Establishment and Rendezvous Vehicle

General Propulsion System Description

I. Energy Requirements

A. Total Ideal Velocity Increment Required = 2700 fps

B. Maximum Velocity Increment

1. Increment = 2200 fps

2. Mission: 5 deg plane change

C. Minimum Velocity Increment

1. Increment = 1 fps

2. Mission: Rendezvous

D. Number of Increments

1. Maximum = 4

2. Minimum = 2

E. Maximum Cutoff Impulse Velocity Uncertainty = 0.5 fps

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.1

b. Absolute Value = 12,000 lb

[redacted]

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TABLE I-30

(Continued)

2. Tolerance

   a. Engine-to-Engine: ± 3 percent

   b. Run-to-Run: ± 1 percent

3. Throttling

   a. Step: None

   b. Continuous: None

4. Number of Restarts: 3

III. Propellants

A. Composition: Mixed Oxides of Nitrogen/Monomethylhydrazine

B. Mixture Ratio

1. Nominal: 2.4 (O/F)

2. Tolerance: ± 0.5 percent

C. Specific Impulse

1. Reference Engine Parameters

   a. Mixture Ratio: 2.4 (O/F)

   b. Chamber Pressure: 150 psia

   c. Expansion Ratio: 25

2. Nominal Engine Specific Impulse at Reference Conditions: 317 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid for engine start.

TABLE 1-30

(Continued)

2. Number of Zero Gravity Engine Starts: 4

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth Vicinity

2. Storage Time: 1 hr. to 1 day (depending on landing requirements)

3. Propellant Temperature Limits

a. Mixed Oxides of Nitrogen

(1) Lower: Freezing (-76 F)

(2) Upper: Propellant vapor pressure and density shall not exceed limits of propellant tanks and engine.

b. Monomethylhydrazine

(1) Lower: Freezing (-63 F)

(2) Upper: Vapor pressure and density shall not exceed propellant tank or engine limits.

C. Component Design Restrictions: Design for operation in earth vicinity space environment or provide protection from the environment.

F. System Purging Requirements:

1. Number of Purges: 3

System Component Requirements

I. Airframe and Propellant Tanks

A. Propellants

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[redacted]

TABLE 1-30

(Continued)

1. Propellant Description

   a. Propellants: Mixed Oxides of Nitrogen/Monomethylhydrazine

   b. Nominal Mixture Ratio: 2.4 (O/F)

2. Useable Propellant Weight: 28,000 lb

3. Reserve Propellant Weight

   a. Flight Performance: 280 lb

   b. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 8 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purposes of Pressurization: Provide energy for expelling propellants from tank into combustion chamber

B. Gas Volume in Propellant Tank

1. Increments: 4

[redacted]

TABLE 1-30

(Continued)

E. Environment

1. Storage

a. Time: 1 Day Maximum

2. Thermal: Earth Vicinity

III. Engine System

A. Propellant Description

1. Propellants: Mixed Oxides of Nitrogen/Monomethylhydrazine

3. Mixture Ratio

a. Nominal: 2.4 (O/F)

b. Tolerance: ± 0.5 percent

B. Thrust

1. Nominal: 12,000 lb

2. Tolerance

a. Run-to-Run: ± 1 percent

b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Pressurized Gas

D. Specific Impulse

1. Nominal: 317 sec

G. Throttling Requirement

1. Step: None

2. Continuous: None

I. Environment: Earth Vicinity

[redacted]

RECOMMENDATIONS FOR FUTURE INVESTIGATION

The current NASA investigation was directed to be a broad study to indicate the general propulsion requirements for space missions. Thus all aspects of the investigations were not all-inclusive and have in themselves revealed areas needing additional effort. In the following section the recommendations for further analytical and experimental investigation of space propulsion systems and requirements are presented.

The investigation areas have been grouped into three general categories: Manned Space Mission investigations to consider those aspects relating to vehicles and propulsion for manned missions; Maneuver oriented space propulsion investigations to formulate propulsion requirements and systems for specific areas in the space mission; Space propulsion analytical and design investigations to evaluate in detail specific areas of space propulsion design which the current study has indicated to require further effort. In these three areas, certain specific areas of investigation which warrant additional effort were uncovered and are presented.

MANNED SPACE MISSION INVESTIGATION

From consideration of the life support capsules [illegible] it is evident that for [illegible] space [illegible]

[redacted]

large space vehicles are necessary. This is particularly true for any extended trips such as to Mars or Venus. In view of this requirement, it is recommended that investigations be conducted in the following areas to further define the design and operation philosophy for extended space missions.

Large Booster Vehicle Development

Development of larger booster vehicles will allow the use of larger space vehicles. Vehicles of the Nova class having initial thrusts in the 9 to 12 million pound range permit the use of a single space vehicle for manned lunar landing and return missions. For more extended space voyages, the space vehicle could be built-up in orbit from models transported by several boosters. Further investigation of the design aspects, economic considerations, and operational requirements should be conducted.

Earth Orbit Rendezvous and Vehicle Buildup

The assembly of space vehicles in an earth orbit will allow the buildup of large space vehicles sufficient for manned flight. Rendezvous guidance and propulsion techniques as well as methods of vehicle assembly should be investigated.

Planetary and Lunar Rendezvous and Vehicle Buildup

Space vehicles can be built up not only in the vicinity of the earth, but in orbits around, or on the surface of, the Moon and planets. A number of vehicles could be sent to the target ahead of the manned vehicles. These could be used for assembly of a return vehicle, furnishing supplies while on the surface of the body, providing vehicle redundancy, or refueling at some point during the trip.

Advanced Space Propulsion System Development

The development of advanced propulsion systems such as the thermonuclear and ion-electrical propulsion systems would considerably reduce the size of the space vehicles required for accomplishing various space missions. This is particularly true for the extended planetary missions. An example in the summary that for a Mars orbit establishment and return mission the nuclear and ion space vehicles were some 27 and 5 percent respectively of the size of the liquid chemical vehicle. The high specific impulses available with these systems would greatly facilitate manned space flight. Investigation of these advanced propulsion schemes should be continued.

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Particulate Radiation Prediction and Protection Systems

The largest single weight item in the life support capsules is the shielding used in protecting the man from particulate radiation. Capsule weight and, thus, space vehicle weight could be significantly reduced. This would be particularly evident in the lunar missions. Through prediction of solar flares and avoiding the Van Allen belts it may be possible to eliminate the shielding for missions of short duration. Where these environmental factors cannot be avoided, advanced shielding methods, using electromagnetic fields for example, may result in a considerable weight savings. Various drugs and treatments might possibly be discovered which would partially "immunize" the astronaut to the effects of the radiation resulting in lower shielding requirements. These methods should be investigated prior to manned flight in any case and offer the possibility of considerable savings over the shielding weights presently envisioned.

Reliability and Abort Considerations for Manned Missions

Manned space missions should be investigated in terms of mission reliability and abort procedures. Both of these features could significantly affect the mission, operational aspects, and propulsion systems. Provisions for return to Earth in case of an abort should be investigated at all points of the missions. The aborts considered can result from minor malfunctions, catastrophic malfunction, and factors not related to the vehicle such as a solar flare. It is possible that an ultra-high reliability escape vehicle can be developed which while perhaps offering low performance will not compromise the mission.

Reliability should also be considered in the overall propulsion system design and mission operation. Some methods of accomplishing a mission may be more reliable than others. The tradeoff between energy requirements and reliability should be investigated.

MANEUVER ORIENTED SPACE PROPULSION INVESTIGATIONS

Earth Escape Phase Propulsion

Analyses of propulsion for the earth escape phase, either for initiation of a lunar or planetary mission, have provided basic criteria which characterize these propulsion systems. These analyses have indicated thrust levels, propellants, restart requirements, cutoff impulse tolerances, etc. The assumption that the space vehicle be launched from an orbit around the earth is inherent in all of the analyses.

Further investigation in this phase should be directed toward operational aspects of the launch from orbit, and toward guidance and trajectory controls considerations. Specific flight trajectories should be established to investigate such areas as positioning or triggering for initiating the space propulsion phase, orbital plane change requirements, and final orbital flight trajectory attitude control for the powered flight. These investigations would provide the final criteria for designing a detailed integrated propulsion system vehicle stage. Many of these requirements mentioned are specific for each mission and would possibly be investigated under a prime contract to develop an Earth-escape-phase stage. However, such an investigation would likely be directed toward a specific mission. A general investigation could consider numerous space missions and establish general propulsion criteria for this Earth-escape phase which would be applicable to numerous missions. Basic review indicates that the J-2 200,000-lb-thrust O_2 H_2 engine, presently under development, would provide highly satisfactory propulsion for this stage.

Orbital Rendezvous

As previously mentioned, orbital rendezvous offers great promise for use in space missions. Review of propulsion requirements for orbital rendezvous indicates several concepts can be used satisfactorily.

Common to all types is the applicability of low thrust. Either a continuously throttleable low-thrust engine system or an intermittent pulsed [illegible] thrust system appears adequate. Other schemes employing fixed low-thrust engine systems could also be employed; however, they might require [illegible] increased propellants.

Review of the analysis conducted and of the available literature indicates that the operational aspect of orbital rendezvous is one prime [illegible]. An investigation in this area is recommended. Concepts such as the number of reserve units, launching order, injection positioning, etc. should be thoroughly investigated.

For example, if the modules could be identical and [illegible] be boosted to orbit, only a single back-up unit might be required. However, if each module is different, a back-up unit for each module would be necessary. In addition, the concept of initially injecting all the modules at [illegible] location in an orbit, or establishing some unique location [illegible] each different orbit (considering such factors as rendezvous [illegible] and propulsion energy) should be analyzed.

In this area, effort should be conducted prior to [illegible] detailed analysis of propulsion systems to accomplish rendezvous. The operational aspects will provide the criteria to determine whether each module must have a rendezvous propulsion system or whether [illegible] space tug system should be employed.

Space Transfer Phase Propulsion

It is recommended that an investigation be conducted to provide propulsion requirements, and propulsion system descriptions, for propulsive maneuvers required during the interspace transfer phase of the space mission. In this space transfer phase, propulsion is required for midcourse corrections, for attitude control, and for terminal correction maneuvers. Applicable analytical trajectory effort and propulsion analyses have been undertaken in a number of separate projects. These have resulted in much useful information concerning the maneuver propulsion requirements. However, the effort toward establishing definite propulsion systems and operational procedures has been limited.

Space vehicles will require a midcourse correction capability. The uncertainty in the description of the physical universe in engineering terms, and the limitations of vehicle control make control mandatory. Investigations of this maneuver should definitely establish the types of guidance systems to be used and their accuracies, as well as the propulsion system limitations. Analyses should determine the number and time of the corrections, and propulsion system features such as thrust level, propellants, cutoff requirements, and general system operation.

Vehicle attitude control will be essential to space missions: 1) to provide specific vehicle orientation prior to major propulsion phases, 2) to alter vehicle orientation during the mission without affecting the trajectory, and 3) to track celestial bodies. The various perturbing forces must be identified and the propulsion requirements determined. Areas in which the different attitude control methods are applicable should be identified.

The terminal correction maneuver is the final correction to provide correct initial conditions for the terminal, destination propulsion. The effects of continuous modification of the trajectory by perturbative factors and deviations from predicted trajectories due to errors in other coast-phase corrections will be negated during this final maneuver.

Propulsion systems for this application can be formulated that will be applicable to a variety of space missions.

In the evaluation of propulsion for the space transfer phase, the resulting systems will be analyzed to determine applicability to other propulsion requirements in the space mission to prevent duplicate propulsion systems. A possible application would be the orbit establishment or the landing maneuver for the mission terminal phase.

Lunar Landing Propulsion Requirements

The current analysis has provided the basic propulsion and mission requirements to establish a lunar landing, take-off, and return mission. Also, lunar orbital and return mission criteria have been established. However, the effort conducted has indicated certain aspects which should be subjected to further analyses. Effort to minimize engine control and throttling requirements should be conducted based on a trajectory mission analysis study. Schemes to improve the hovering/lateral location operation phase should be investigated further to possibly develop schemes for minimized propulsion requirements for this phase.

Analytical design effort for the lunar mission should be conducted in two main areas. The first area would be design analysis of propulsion subsystems based on the presently formulated propulsion system requirements. The second would be analytical mission/trajectory investigations to attempt to further minimize propulsion requirements, i.e., engine control and throttling.

The lunar mission design analyses could be conducted jointly by a propulsion and vehicle contractor to investigate propulsion systems and their integration with vehicle stage designs for the Lunar mission. This effort would evaluate design details, such aspects as engine packaging, advantages of different propulsion configurations, dynamic analysis of the stage to determine control requirements, analysis of possible damage resulting from a non-zero velocity impact.

The analytical investigation would be conducted to extend the present mission/trajectory analysis in order to minimize the propulsion system requirements. Methods of landing either direct or with an intermediate orbit which would negate the requirement for engine throttling would be attempted. Investigation of trajectories to minimize the optimum thrust-to-weight, to minimize the propulsion system components would be an additional phase of this analytical study.

__Planetary Orbit Establishment__

Present analysis of planetary orbit establishment propulsion has established general requirements. However, the analysis should be extended to determine if a simple basic scheme for establishing a planetary orbit can be achieved which would minimize the control requirements both from the propulsion and guidance/communication standpoint. The interrelation of the interplanetary orbit phase with the midcourses and terminal space correction propulsion requirements, indicate that such an analysis would be undertaken after investigation of space transfer propulsion. It is anticipated that the detailed studies of the orbit establishment phase based on refinement of the space correction propulsion and trajectory control can provide a simple scheme which will provide planetary orbit establishment with a minimum of control and readjustment. Additional trajectory analyses should be initiated to investigate some trajectory details such as the influence of the moons of Mars on approach trajectories and the orbit inclinations resulting from various transfer maneuvers.

Planetary Landing and Return Propulsion

Review of the aspects of a planetary landing mission indicate that initial landing missions undertaken will be based on powered-retrothrust landing rather than an atmosphere drag scheme, because of the lack of data on the planetary atmospheric conditions. An analysis of propulsion requirements for planetary landing should be undertaken to examine special propulsion criteria required. The basic velocity requirements have been established. Based on the lunar mission landing studies, concepts of landing with intermediate surveillance orbits have been formulated.

Present analyses have been primarily concerned with Earth/planet transfers. The reverse missions, initiated at great distances from the Earth, should be investigated. For either planetary-orbit initiated or planetary-takeoff booster vehicles, the propulsion requirements, trajectory accuracy, and operational characteristics resulting from the foreign planet operation should be described.

The landing of a space vehicle on the Earth after a long return space voyage should be analyzed for propulsive and operational requirements. Accuracy requirements for terminal corrections which prepare the vehicle for aerodynamic re-entry should be evaluated, and tradeoffs between re-trothrust and Aerodynamic landings analyzed to determine the desirable single or combination system.

SPACE PROPULSION SYSTEM DESIGN AND OPERATIONAL INVESTIGATION

Cryogenic-Propellant Space Propulsion System Design

Analysis conducted has indicated the desirability of high-energy cryogenic liquid propellant systems in the Earth space missions. However, the analysis has also shown that the deep space missions which require a coast phase of 1 to 2 years (or longer) storable liquid propellants might achieve performance comparable to cryogenic liquid propellants. The major problem area in storage of liquid cryogenic systems is the heat transfer by conduction within the vehicle system itself.

A propulsion vehicle design study to investigate specific system designs and the heat transfer problems involved should be undertaken to assure the feasibility of using the cryogenic liquid propellant systems for long space missions. The detailed design investigation of conduction heat transfer would provide data to determine the degree of performance advantage of the liquid cryogenic system.

Data from current work which present basic boosters and space stage criteria such as sizes, thrust levels, etc. would provide the background for the detailed designs of these space stages. The criteria that would be investigated would be: the insulation weight vs propellant boiloff; methods of supporting the tanks to minimize conduction heat transfer; propellant feed line concepts to minimize conduction heat transfer; attitude control systems to minimize solar radiation heating.

Investigation of meteoroid protection methods for the propulsion system and stage design would also be included. The "Whipple meteor bumpers" appear to be the best system at the present. Additional investigation is needed to provide information sufficient for over-all system final design.

The analytical and design investigation for space propulsion stage systems could be a joint effort of a propulsion and vehicle contractor. This basic design study would provide concepts and detailed design information for the propulsion and the space vehicle design aspects.

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Space Engine Enviromental Design

Propulsion system design and operation aspects should be investigated further from the standpoint of environmental effects. Studies have been conducted in a number of separate areas and comprehensive study of these features relating directly to the engine system would be of great benefit. Multiple restarts in vacuum, valve leakage over long periods of time, heat balance in the engine, and the subsystem designs are areas needing investigation to establish final design criteria.

Propulsion System Operating Parameters

Propulsion system parameters in this study have been selected from a consideration of previous Rocketdyne studies. Methods for estimating optimum values of several parameters have been presented. Investigations of additional parameters, description of influencing factors, selection of optimum values, and indications of the tradeoffs involved in off design operation would be beneficial.

Rocket Engine Exhaust Investigation

The operation of rocket engines in vacuum presents the possibility of several problems. Jet spreading during vacuum operation may create heating problems in the vehicle boattail or on the propellant tanks. Investigations would consider the actual geometry of the jet spread, the heat transfer mechanisms, and protective insulation schemes.

The exhaust jet may create problems in landing in that it may interfere with guidance and vision. The impact of the jet on the surface of a body such as the Moon may create dust and erosion problems. Investigations would be concerned with infrared flame radiation, guidance operation, jet spreading, and jet erosion.